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A USER'S GUIDE TO THE LANGLEY 16-FOOT TRANSONIC TUNNEL COMPLEX

Revision 1

Staff of the Propulsion Aerodynamics Branch

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INTRODUCTION

This document describes in considerable detail the operational characteristics and equipment associated with the Langley 16-Foot Transonic Tunnel Complex which is located in Buildings 1146 and 1234 at the Langley Research Center. This complex consists of the 16-Foot Transonic Wind Tunnel, the Static Test Facility, and the 16- by 24-Inch Water Tunnel research facilities. The 16-Foot Transonic Tunnel is a single-return atmospheric wind tunnel with a 15.5 foot diameter test section and a Mach number capability from 0.20 to 1.30. The emphasis for research conducted in this research complex is on the integration of the propulsion system into advanced aircraft concepts. In the past, the primary focus has been on the integration of nozzles and empennage into the afterbody of fighter aircraft. During the last several years this experimental research has been expanded to include developing the fundamental data base necessary to verify new theoretical concepts, inlet integration into fighter aircraft, nozzle integration for supersonic and hypersonic transports, nacelle/pylon/wing integration for subsonic transport configurations, and the study of vortical flows (in the 16- by 24-Inch Water Tunnel).

The purpose of this paper is to provide a comprehensive description of the operational characteristics of the research facilities of the 16-Foot Transonic Tunnel Complex, and their associated systems and equipment. It is compiled for the use of the wind tunnel staff and potential facility users. Hopefully, the information will provide a guide for test preparation to the facility users. It is strongly emphasized that this paper should not be employed as the sole source of information to be used in preparation for tests in this facility. As with any other major wind tunnel, modifications to these facilities, their supporting equipment, test hardware, and associated data acquisition and reduction systems, take place on a continuing basis. Direct contact should, therefore, be established with facility personnel prior to the time the model or hardware design is initiated.

Use of trade names or manufacturers' names in this report does not constitute an official endorsement of such products or manufacturers, either expressed or implied, by the National Aeronautics and Space Administration.

The information contained in this report supersedes NASA TM-83186, A User's Guide to the Langley 16-Foot Transonic Tunnel, published in August of 1981. This updated version was prepared by the staff of the Propulsion Aerodynamics Branch with significant contributions by Ms. L. S. Bangert and Messrs. B. L. Berrier, F. J. Capone, J. R. Carlson, G. T. Carson, A. M. Ingraldi, T. T. Kariya, L. D. Leavitt, C. E. Mercer, D. H. Neuhart, O. C. Pendergraft, Jr., and R. J. Re.

SECTION I - Description of Facility

A. <u>Nominal Test Range.</u>- The Langley 16-Foot Transonic Tunnel, located in Building 1146 and shown in figure I-1, is a closed circuit, single-return, continuous-flow atmospheric tunnel. The test medium is air with an air exchange for cooling. The normal testing range consists of Mach numbers up to 1.30 and angles of attack up to 25 . Speeds up to Mach 1.05 are obtained with the tunnel main drive fans; speeds above Mach 1.05 are obtained with combination of the main drive fans and test-section plenum suction. The slotted octagonal test section nominally measures 15.5 feet across the flats. The useable test section length is 22 feet for speeds up to Mach 1.0 and 8 feet for speeds above M = 1.0.

B. <u>Types of Testing Conducted</u>.- The facility is used for studies of aerodynamic characteristics, flow simulation, and flow analysis of aircraft configurations at transonic speeds. The aerodynamic characteristics are obtained on both powered (propulsive) and unpowered models with flow simulation (engine/nacelle) including inlet (flow through) and exhaust (cold flow) studies. Analysis of the flow field surrounding a model can be made by utilizing either laser velocimetry or pressure survey measurement techniques.

C. <u>Historical Data.</u>- The original wind tunnel was designed and built in the expansion period of the National Advisory Committee for Aeronautics (NACA) prior to World War II and was put into operation in November 1941 as the Langley 16-Foot High-Speed Tunnel. The original tunnel had a closed circular test section 16 ft. in diameter and was driven by two 8000 hp electric motors mounted in the return passage. The drive motors were directly coupled to counter-rotating fans operating in tandem. The wind tunnel was cooled by air exchange, and the maximum test section Mach number was 0.71.

The facility was designed and used primarily for aircraft engine cooling and cowling tests and for the investigation of aerodynamic characteristics of full-scale propellers. The need for increased air speed capability led to a major revision of the original tunnel prior to 1950. During this time, the NACA had developed the slotted-wall transonic test section and this type of test section was selected for the revised tunnel.

The major revisions which were incorporated in this modification consisted of a 60,000 hp drive system, an octagonal slotted test section, air filters in the air exchange system, acoustical treatments and a new control room. The repowered facility, designated the "Langley 16-Foot Transonic Tunnel," was placed in operation on December 6, 1950 and had a maximum Mach number of 1.10. Work planned for the repowered tunnel pertained to the extension of propeller aerodynamic data to supersonic speeds and to extensive study of jet effects on aircraft performance. Reference I-1 contains information on this tunnel configuration including the slot shape development up to slot shape 18 which was used up to 1961.

The most recent repowering of the Langley 16-Foot Transonic Tunnel required primarily the provision of a 36,000 hp compressor capable of removing about 4.5 percent of the tunnel mass flow by way of the test section wall slots and surrounding plenum with the removed air being exhausted to the atmosphere. The wind tunnel with this test section air removal system was placed in operation on March 9, 1961. The maximum test section Mach number is 1.30. The calibration of the facility with air removal was initially conducted from March to June of 1961 during which time slot shape 26 was developed. Further refinements to slot shape 26 were made in 1963 and 1965 resulting in slot shape 29 which is the present slot configuration. The results of these three calibrations are contained in reference I-2.

In 1977, during a major rehabilitation of the facility, new fan blades, new tunnel controls and associated control room equipment, and an on-site computer were installed. However, no changes were made to the test section. A check calibration was performed in 1977 and as expected, there was no change in the flow characteristics of the facility.

From mid 1989 to 1990, significant modifications were made to the 16-Foot Transonic Tunnel. These included installation of a new model support system, new fan blades, a catcher screen installed on the first set of turning vanes, new process controllers (tunnel, strut, air) and associated control room equipment, and a new computer system. In addition, a semi-span test apparatus and a model preparation room were also provided.

D. Description of the Facility Components.- The Langley 16-Foot Transonic Tunnel is a single-return atmospheric wind tunnel having a slotted test section. An exterior view of the facility is shown in figure I-1, a phantom view (without air removal equipment) in figure I-2, and a schematic diagram in figure I-3. Starting at Tunnel Station 0 (Tunnel Stations given in feet), the major components are the quiescent chamber, entrance cone, test section, diffuser, power section, return passage, and air exchange section. There are four sets of turning vanes located at the respective 90° elbows in the tube and two antiturbulence screens, one in the air exchange section and another in the quiescent chamber. The length of the tunnel circuit along the center line is 930 ft. and the maximum inside diameter in the quiescent chamber is 58.0 ft. The test section is a regular octagonal cylinder having a cross-sectional area slightly less than that of a

16-foot diameter circle. The test section air removal equipment is located outside the tunnel between the diffuser and the return passage. A description of some of these components follows, and a more complete description can be found in reference I-2.

D.1. Quiescent Chamber and Entrance Cone.- The dimensions of these components provide a contraction ratio of 13.31 between the large end of the wind tunnel and the test section. At test section Mach numbers above 1.0, the average air speed in the quiescent chamber is 54 ft/sec. This low velocity region downstream of the antiturbulence screen permits further decay of residual turbulence prior to acceleration of the airstream through the entrance cone and into the test section. The entrance cone incorporates a transition from circular to octagonal and includes a slowly converging accurately finished entrance liner which terminates at the upstream end of the test section.

D.2 Test Section.- The test section is an octagonal cylinder vented to a surrounding plenum through slots at the corners of the octagon. A schematic of the test section is given in figure I-4 and photographs are presented in figure I-5. The test section and diffuser entrance, which are the portions of the wind tunnel having variable geometry, extend from Tunnel Station 107 to 154 as indicated in figure I-4. The cross section at Tunnel Station 107 is a closed regular octagon having an area of 199.15 ft.^{2}. The feature which gives the wind tunnel transonic capability is the venting to the plenum. In this tunnel, the vents are eight longitudinal slots located at the intersections of the wall flats. The plenum is a sealed tank 32 ft. in diameter, which encloses the test section and diffuser entrance.

The test section is made up of eight longitudinal flats symmetrically located about the tunnel center line. The surface of each flat is a continuous steel plate from Tunnel Station 107 to 154 with the maximum width being 6.66 ft. at the upstream station. Each flat is rigidly supported by a main truss and strong-back between Tunnel Stations 114 and 138 and by a second truss and strong-back between Tunnel Stations 140 and 154. The long steel plates which form the walls are attached rigidly to the tunnel structure between Tunnel Stations 107 and 110. The plates have flexural regions centered at Tunnel Stations 112 and 139 and a sliding joint at Tunnel Station 154, where there is a sealed traverse gap which varies with wall divergence. The region where the the tunnel walls are flat, therefore, extends from Tunnel Station 114 to 138. When the test section walls are diverged, bending takes place between Tunnel Stations 110 and 114 and divergence is measured as the angle between the tunnel center line and the rigid portion of the wall from Tunnel Station 114 to 138.

The test section slots, located at the intersections of the wall flats, are eight longitudinal openings generally parallel to the tunnel center line which provide vents between the test section airstream and the plenum which surrounds the test section. The slot width or opening is nominally zero between Tunnel Stations 107 and 107.5. Each slot starts at Tunnel Station 107.5 and extends to Tunnel Station 154, but is closed by the diffuser entrance vanes between Tunnel Stations 141.5 and 154.

D 3. Diffuser.- The overall diffuser extends from the downstream end of the test section to the power section. The function of the diffuser is to decelerate the airstream after its passage through the test section and thereby to convert as much as possible of its kinetic energy into pressure energy.

D.4. Power Section and Drive Fans.- The wind-tunnel tube at the drive end has a constant diameter of 34 ft., and the power section includes two

90° elbows which incorporate, respectively, the first and second set of turning vanes (fig. I-3). A catcher screen is located on the upstream side of the first set of turning vanes. The arrangement of the major components of the tunnel drive end is indicated in figures I-2 and I-3. The two main drive motors, housed outside the tunnel, are each connected directly to one of the drive fans through a shaft about 60 ft. long. The enclosure which houses the fan hubs, jack shafts, and bearing pedestals for both units has a streamlined shape with a maximum diameter of 20 ft. in the vicinity of the fan stations.

The two main drive electric motors are of the wound rotor type and each is rated for continuous operation at 23,000 hp at a rotational speed of 340 rpm, for 2 hours of operation at 30,000 hp at 366 rpm and for 1/2 hour operation at 34,000 hp at 372 rpm. The rotational speed of the motors is controlled by a modified Kramer system which permits essentially continuous variation of speed from 60 rpm to 372 rpm.

The drive fans constitute a two-stage axial-flow compressor having two sets of counter-rotating blades with no stator blades. The fans are 34 ft. in diameter less 0.2 in. radial clearance between the blade tip and the tunnel wall. The fan blades are made of laminated spruce. The upstream fan has 25 blades and the downstream unit has 26 blades. The blades have Clark Y airfoil sections.

D.5. Return Passage.- The return passage upstream of the air exchange section is a large conical diffuser and downstream, a cylinder. The primary function of the return passage is to duct reenergized air from the power section through the air exchange section back to the quiescent chamber. Air velocities throughout the return passage are too low to yield much pressure recovery in the diffuser portion of the return passage. The cylindrical portion has two 90° elbows which incorporate the third and fourth sets of turning vanes.

D.6. Air Exchange Section.- The air exchange section serves to cool the wind-tunnel airstream and to provide scavenging of exhaust gases for engines if used during investigations in the test section. All the energy expended through the main drive fans is eventually converted into heat which elevates the airstream temperature. With no cooling of the wind tunnel during operation at high power, the airstream temperature would increase rapidly to a dangerous level. The Langley 16-Foot Transonic Tunnel was designed primarily for investigation of propulsion system effects on airframe aerodynamic characteristics. Investigations of this type may entail the operation in the test section of real engines emitting hot and toxic exhaust gases which must be scavenged continuously from the wind-tunnel airstream. The process of cooling by air exchange consists of exhausting a part of the wind-tunnel airstream which has become heated and by replacing the heated air with cool ambient air. The air exchange section which performs this function is shown in figures I-2 and I-3.

D.7. Antiturbulence Screens.- Two antiturbulence screens, each composed of a single layer of square mesh woven wire, are installed one in the air exchange section and the other in the quiescent chamber (fig. I-3). In addition to reducing turbulence, the screen in the air exchange section also increases the effectiveness of the air exchange by creating a slight pressure drop between exhaust and intake.

D.8. Test Section Air Removal System.- Figures I-1, I-3, and I-6 show the general arrangement of the test section air removal equipment with respect to the wind tunnel proper. Basically, this is a large motor-driven axial-flow compressor which removes low energy air (up to 4.5 percent of test section mass flow) from the plenum surrounding the test section and discharges this air to the atmosphere.

Test section air removal is beneficial primarily for the attainment of low supersonic speeds in a transonic wind tunnel. In applying test section air removal, the compressor is sized to pump stagnant air from the plenum and to exhaust it to the atmosphere. The pressure difference and compressor mass flow are determined by the test section Mach number and air state. The compressor pumping redirects the slot mixing losses and some of the tunnel skin-friction loss because boundary-layer air is removed. The tunnel main drive then, aided by this quasi-boundary-layer control, is required to overcome the normal friction and turbulence losses of the wind tunnel and the test section terminal shock. Thus, to achieve a specified low supersonic speed in a slotted wind-tunnel test section, less total power is required when applied partly in the main drive and partly in a test section air removal system than when all the power is applied through the main drive. The air removal compressor is a nine-stage axial-flow compressor with a 3.33 total pressure ratio. The maximum rotational speed is 2290 rpm and the inlet volume flow is 956,000 ft.^{3}/min. The compressor is driven by a wound rotor induction through a speedup gear. The motor rating is 36,000 hp at 552 rpm. Speed control is obtained by a slip regulator incorporating a brine tank rheostat.

The flow of air from the test section to the compressor is controlled by a 10-ft. diameter hydraulically actuated butterfly valve located in the air removal duct just outside the test section plenum as indicated in figure I-6. This valve is automatically controlled to set Mach numbers between 1.05 and about 1.25 as the main drive are held at constant speed. The main drive speed is then increased for Mach numbers greater that 1.25.

E. Flow and Operational Characteristics

E.1. Operating Variables.- Experience acquired in operation of the wind tunnel with test section air removal have established the techniques for obtaining any preselected value of Mach number up to 1.30. The setting of a particular Mach number is a function of the speed of the main drive fans and compressor, positions of the plenum and surge control valves, and values of test section wall divergence. Varying wall divergence is used to maintain a zero axial static-pressure gradient in the test section. Mach number is only a function of main drive fan speed and wall divergence for Mach numbers less than 1.05. For Mach numbers greater than 1.05, Mach number is a function of main drive fan and compressor speed, plenum valve position, wall divergence, and wind tunnel airstream dew point. Experience has shown that although moisture content of the airstream did not have a large effect on aerodynamic data (ref. I-3 and unpublished data), it did have some effect on the axial gradient of static pressure in the test section at supersonic speeds. Although the gradient does not effect the mean value of Mach number in the test section, a static-pressure gradient can exert horizontal buoyancy on a model and may cause an error in drag measurements.

E.2. Wind-Tunnel Operational Characteristics.- Typical centerline Mach number distributions for slot shape 29 are shown in figure I-7. The variation with Mach number of Reynolds number per foot is presented in figure I-8 for the corresponding range of operating air stagnation temperature. The variation of static pressure, dynamic pressure, and equivalent pressure altitude with Mach number in the test section is presented in figure I-9. Further details of the calibration of the 16-Foot

Transonic Tunnel are presented in reference I-2. Typical power requirements for the wind tunnel are shown in figure I-10.

E.3. Airstream Turbulence.- During the wind-tunnel calibration of 1950 - 1951, with slot shape 18 (ref. I-1) and on several occasions prior to the use of test section air removal and before an antiturbulence screen was installed in the quiescent chamber, attempts were made to measure airstream turbulence. Fluctuating flow angle was measured with a three degree cone meter, but interpretation of the data was uncertain. If isotropic turbulence can be assumed, those early results indicated that the stream angle fluctuation expressed in radians ranged from about 0.003 to 0.008. Data from reference I-4 are presented in figure I-11.

No comparable measurements of fluctuating flow have been made since operation with test section air removal was started with slot shape 29. However, subsequent to the installation of the antiturbulence screen and slot shape 29, measurements have been made of the length of run of laminar flow on a highly polished 10° cone, and some of the results are reported in reference I-5. These measurements were made in the Langley 16-Foot Transonic Tunnel at Mach numbers up to 1.30 by the same investigators using the same techniques and equipment as for the other investigations reported reference I-5. Analysis of the flow transition data presented in reference I-5 and figure I-12 indicates that longer runs of laminar flow were obtained in Langley 16-Foot Transonic Tunnel than in the other wind tunnels investigated. A long run of laminar flow is interpreted to indicate a low level of airstream turbulence.

F. Guidelines For Model Location And Size

F.1. Model Location.- Consultation with Propulsion Aerodynamics Branch personnel is necessary to ascertain both the size and appropriateness of a projected model location in the test section for the range of freestream flow conditions in the test program. For subsonic Mach numbers, the model nose can be located as far forward as Tunnel Station 115. However, the model base should be located forward of Tunnel Station 137 in order to avoid the decrease in Mach number that occurs at about Tunnel Station 137.5 as shown in figure I-7. At supersonic numbers, the model nose should not be located forward of Tunnel Station 128.5 because uniform supersonic flow has not yet been developed in the test section (fig.I-7).

F.2. Model Size.- Generally, model size is dictated by wind tunnel blockage effects at subsonic speeds and boundary-reflected disturbances at supersonic speeds. The maximum cross-sectional area of the model can limit the transonic Mach number at which interference-free data can be obtained. Experimental blockage studies, which have been made in the 16-Foot Transonic Tunnel (ref. I-6) with slot shape 29 (width of 3.9 percent open periphery), indicates that no serious wall interference occurred at subsonic Mach numbers up to M = 0.98 for models having blockage ratios of about 0.0006. These results are presented in figure I-13. These results indicate, for example, that interference-free data can be obtained up to a Mach number of about 0.97 for a model with a blockage ratio of 0.0008 which would have an equivalent diameter of 5.4 in. Similarly, interference-free data could be obtained up to a Mach number of about 0.95 for a model with a blockage ratio of 0.0024 which would have an equivalent diameter of about 9.4 in. In general, no testing is permitted in the 16-Foot Transonic Tunnel between Mach numbers of 0.98 to 1.05 because of the possibility of both subsonic and supersonic wall interference effects which are discussed in reference I-7.

At low supersonic Mach numbers, boundary-reflected disturbances tend to limit the useful test section length. Surface static-pressure measurements made on several cone-cylinder or ogive-cylinder bodies of revolution were used to determine the location of the reflected bow shock which determines the extent of the interference-free region. These results, from reference I-7 and including some unpublished data, are presented in figure I-14. The disturbance length is seen to depend on the strength of the bow shock as affected by the body nose angle and change in nose shape from conical to ogive. At Mach numbers greater than 1.15, aircraft models of reasonable length (70.9 in.) are generally free of boundary-reflected disturbances.

References

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I-2. Corson, Blake W., Jr.; Runckel, Jack F.; and Igoe, William B.: Calibration of the Langley 16-Foot Transonic Tunnel With Test Section Air Removal. NASA TR R-423, 1974.

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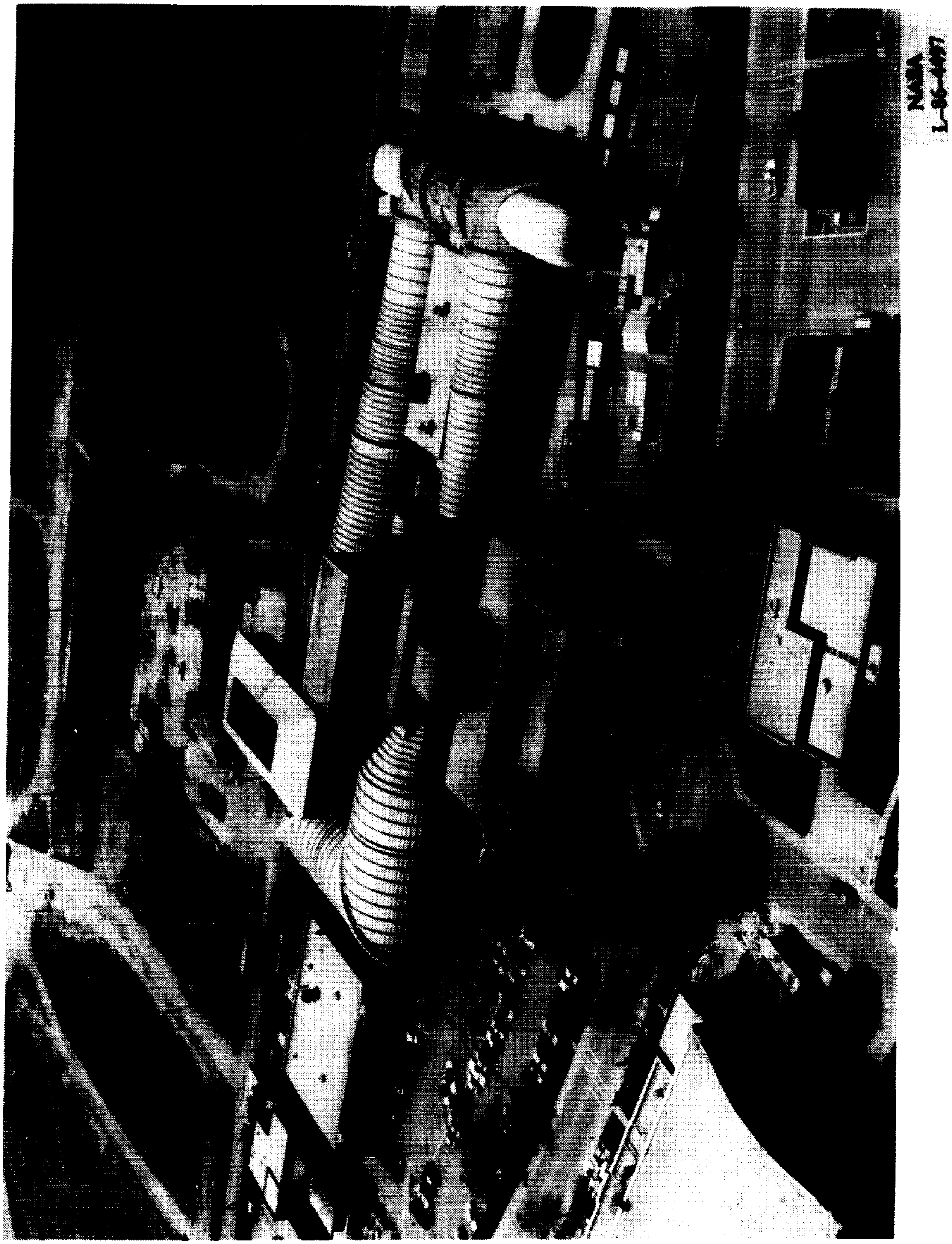

Figure I-1. The 16-Foot Transonic Tunnel at NASA Langley Research Center, Hampton, Virginia.

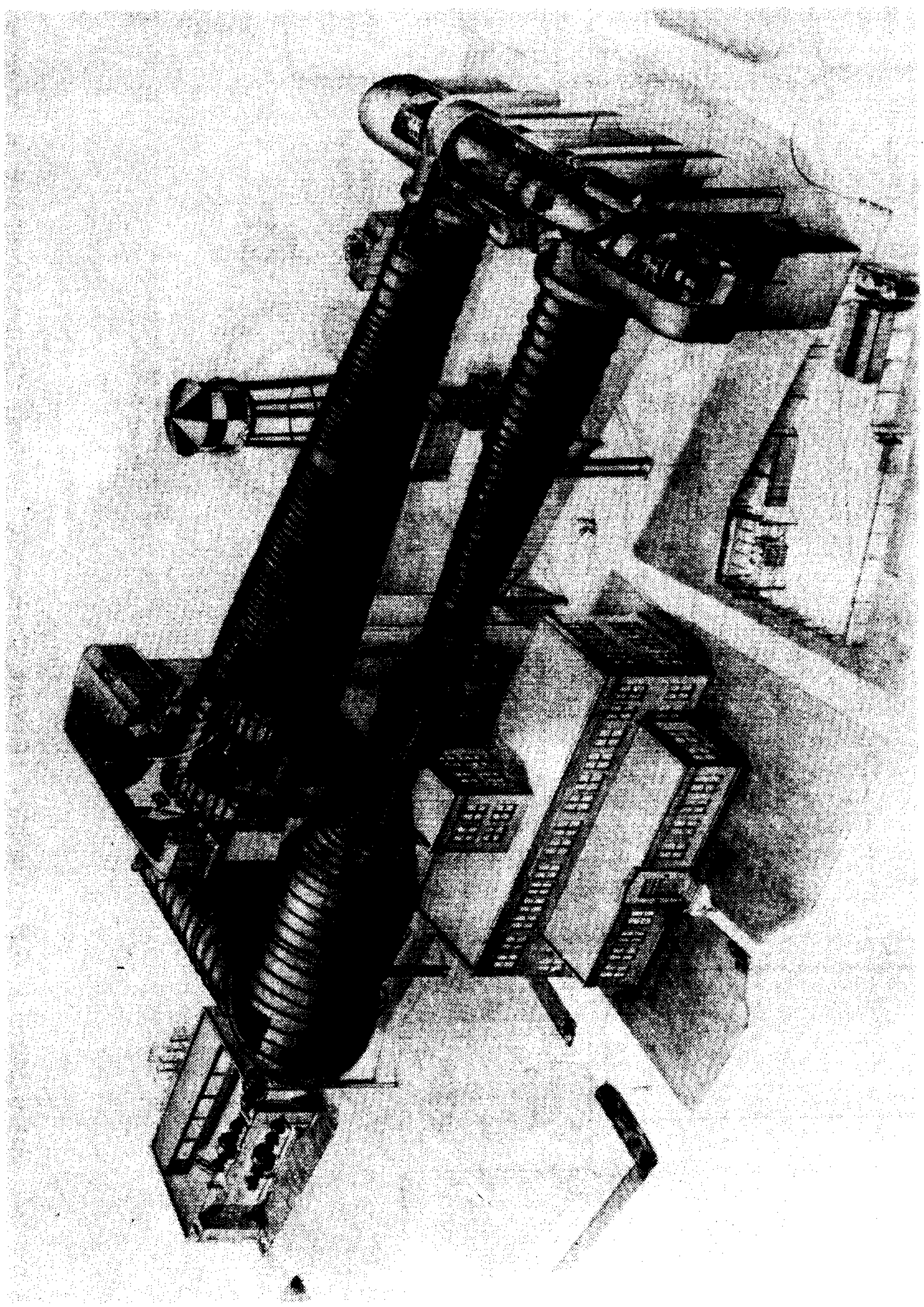

Figure I-2. Phantom view of the 16-Foot Transonic Tunnel (before Air Removal System was added).

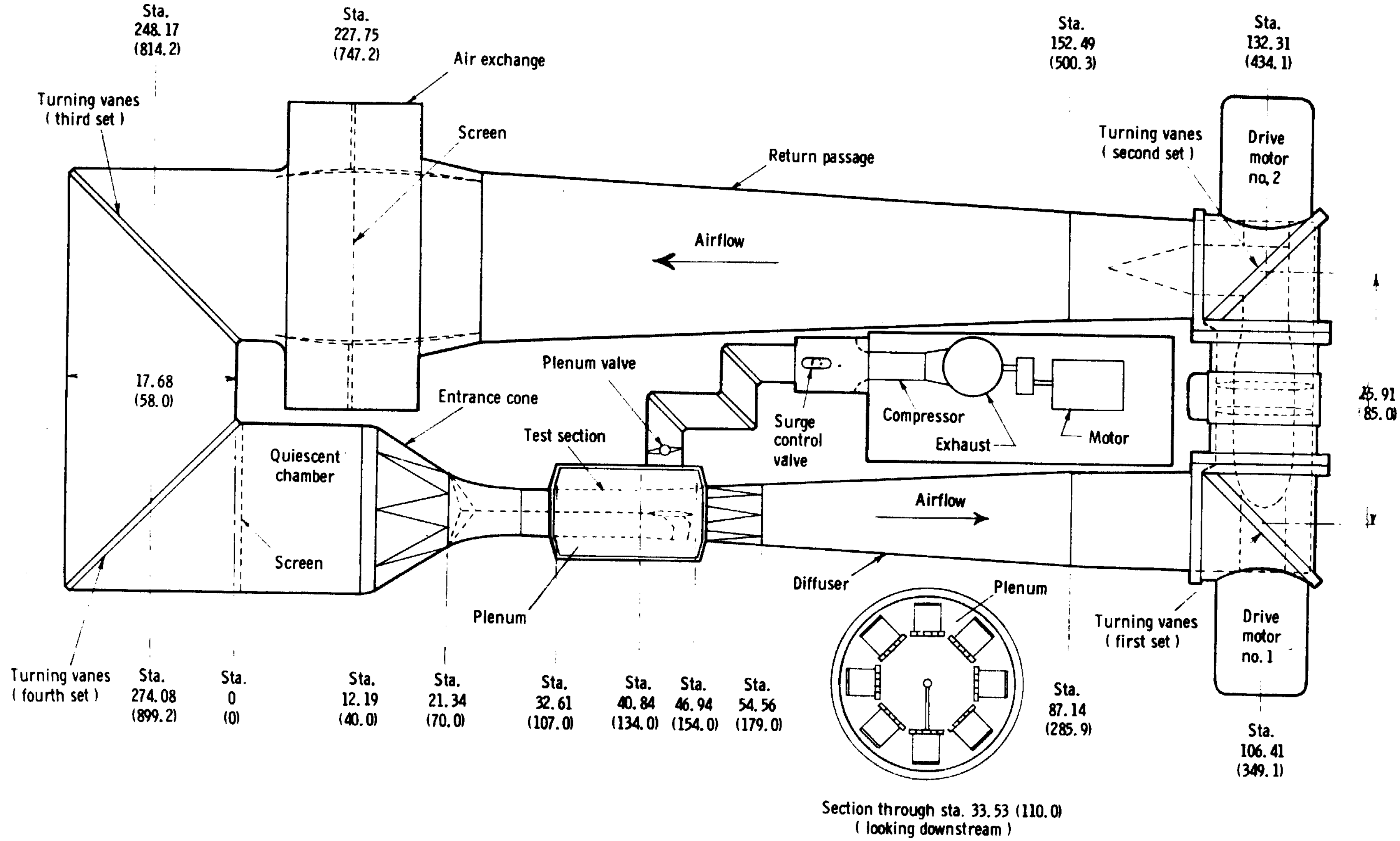

Figure I-3. Arrangement of the Langley 16-Foot Transonic Tunnel. Dimensions are in meters (feet).

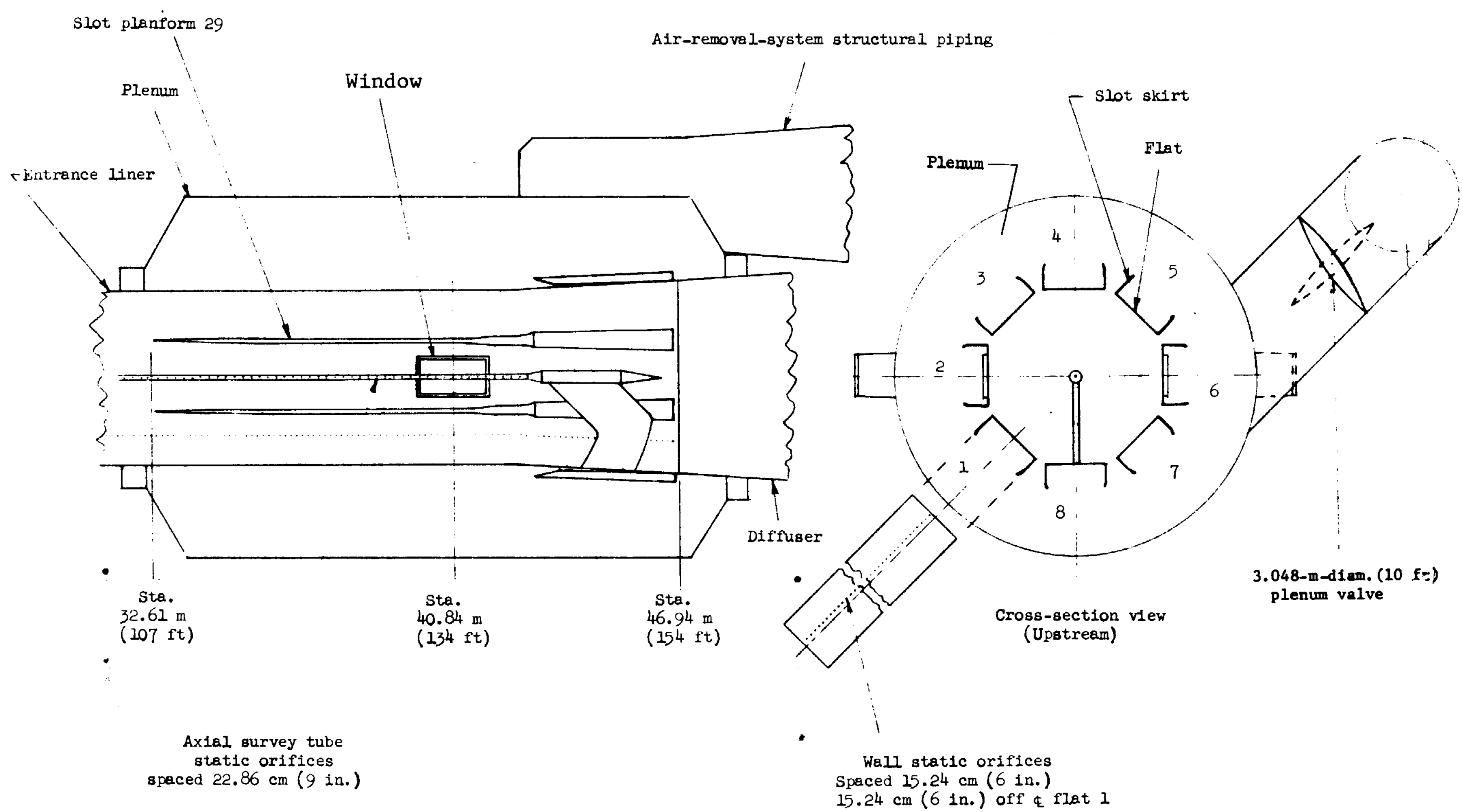

Figure I-4. Schematic arrangement of test section with calibration instrumentation.

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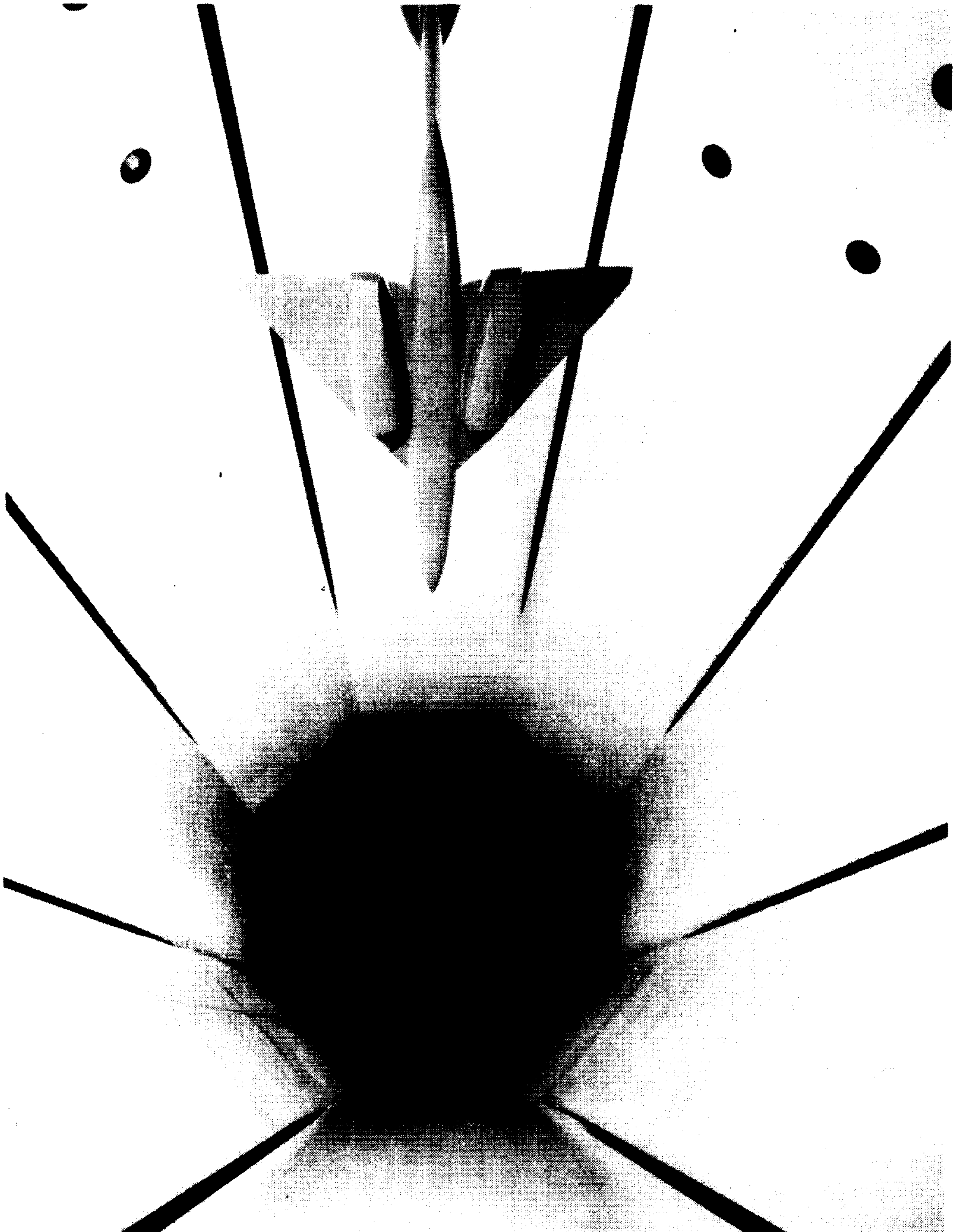

(a) Looking upstream.

Figure I-5. Photograph of test section.

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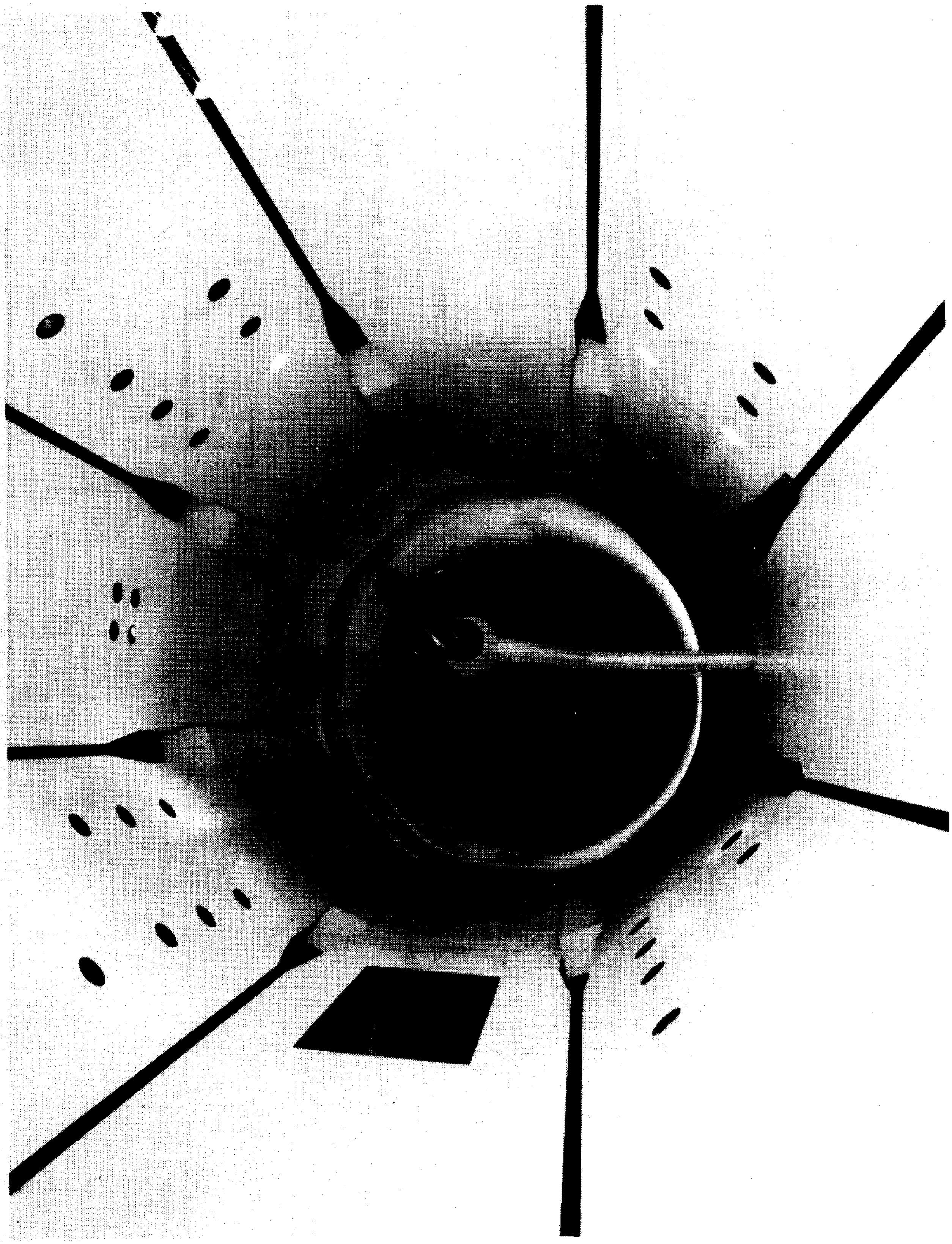


(b) Looking downstream.

Figure I-5. Concluded.

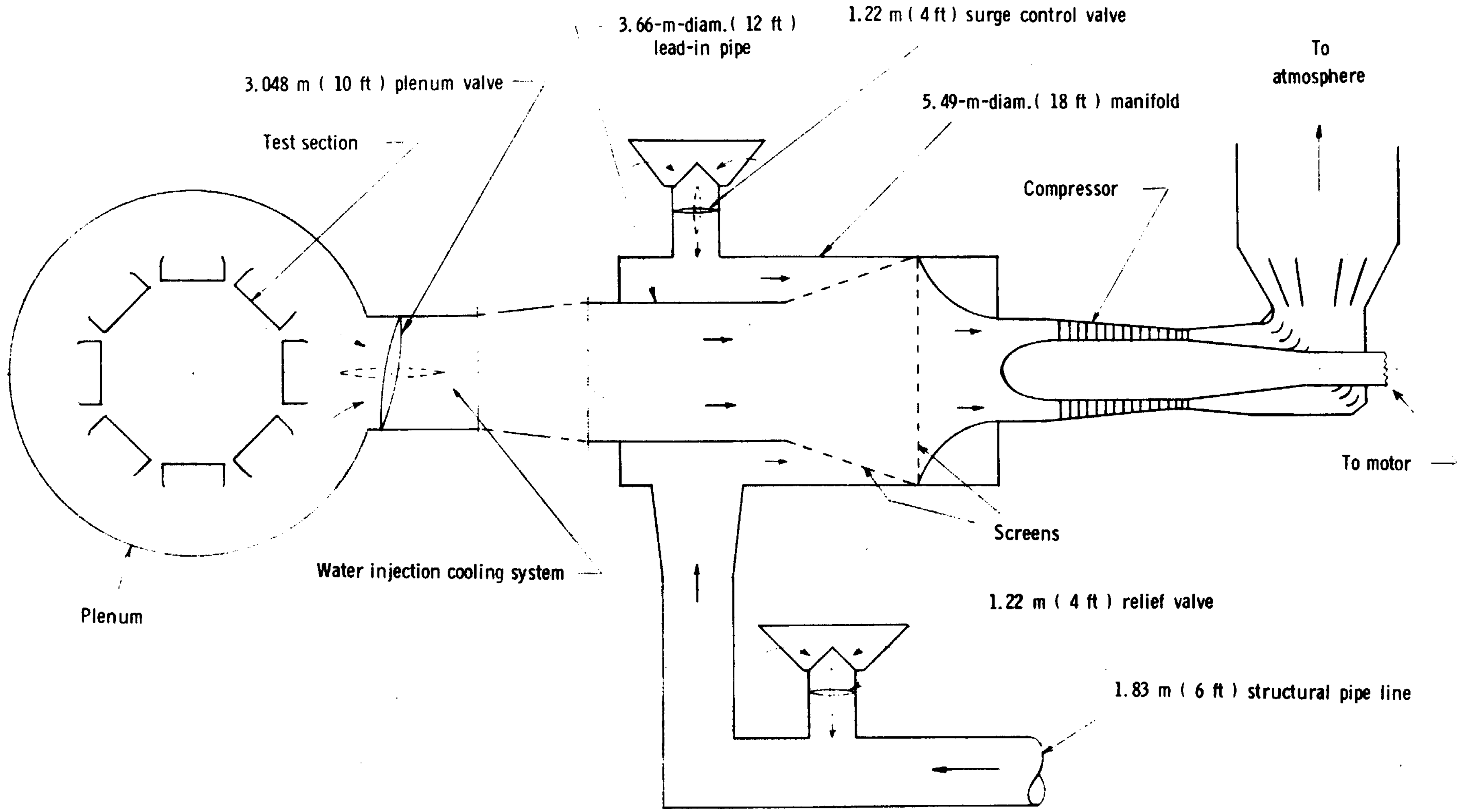


Figure I-6. Schematic arrangement of test section air removal system.

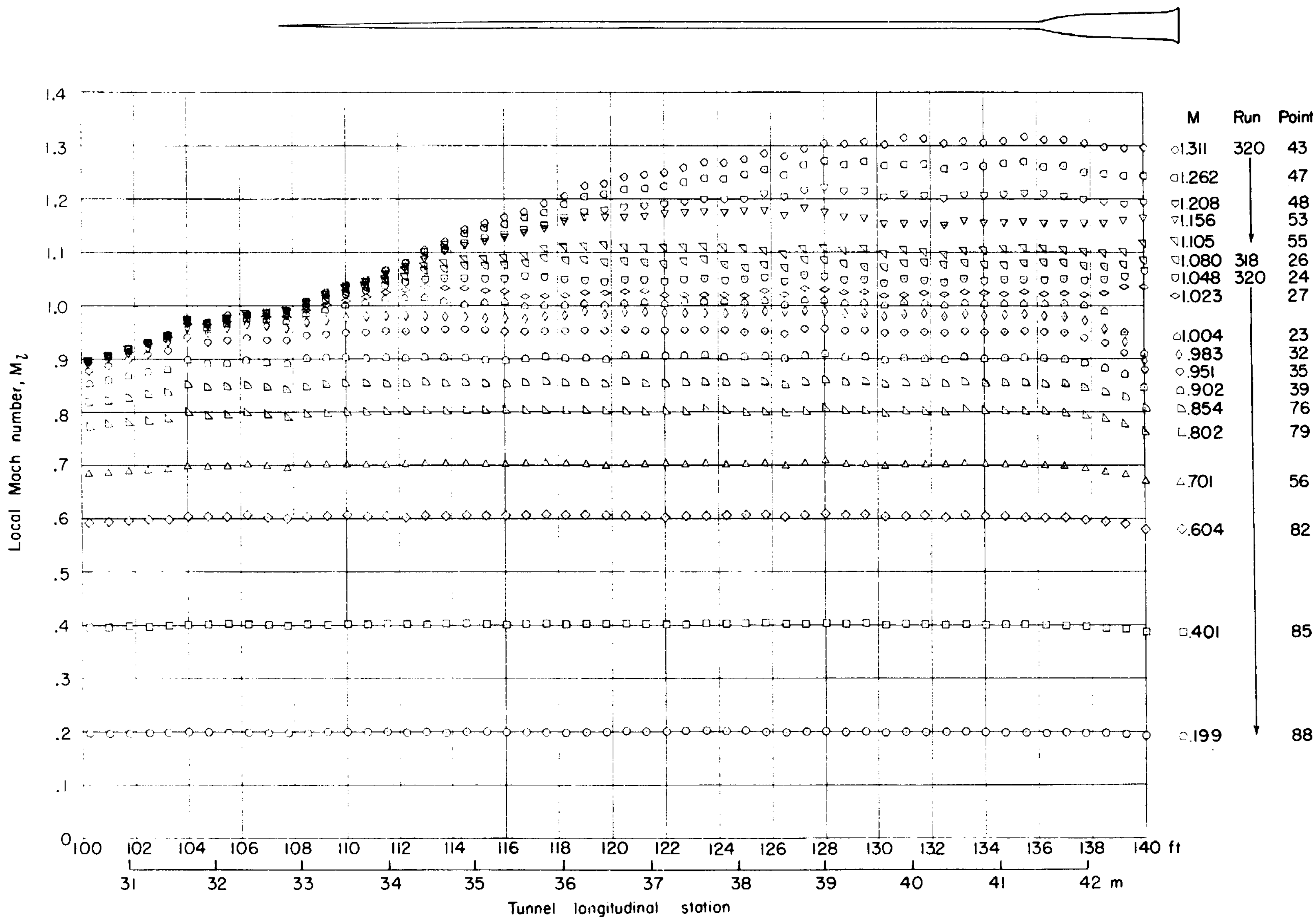


Figure I-7. Longitudinal Mach number distribution along center-line of the test section for slot planform number 29

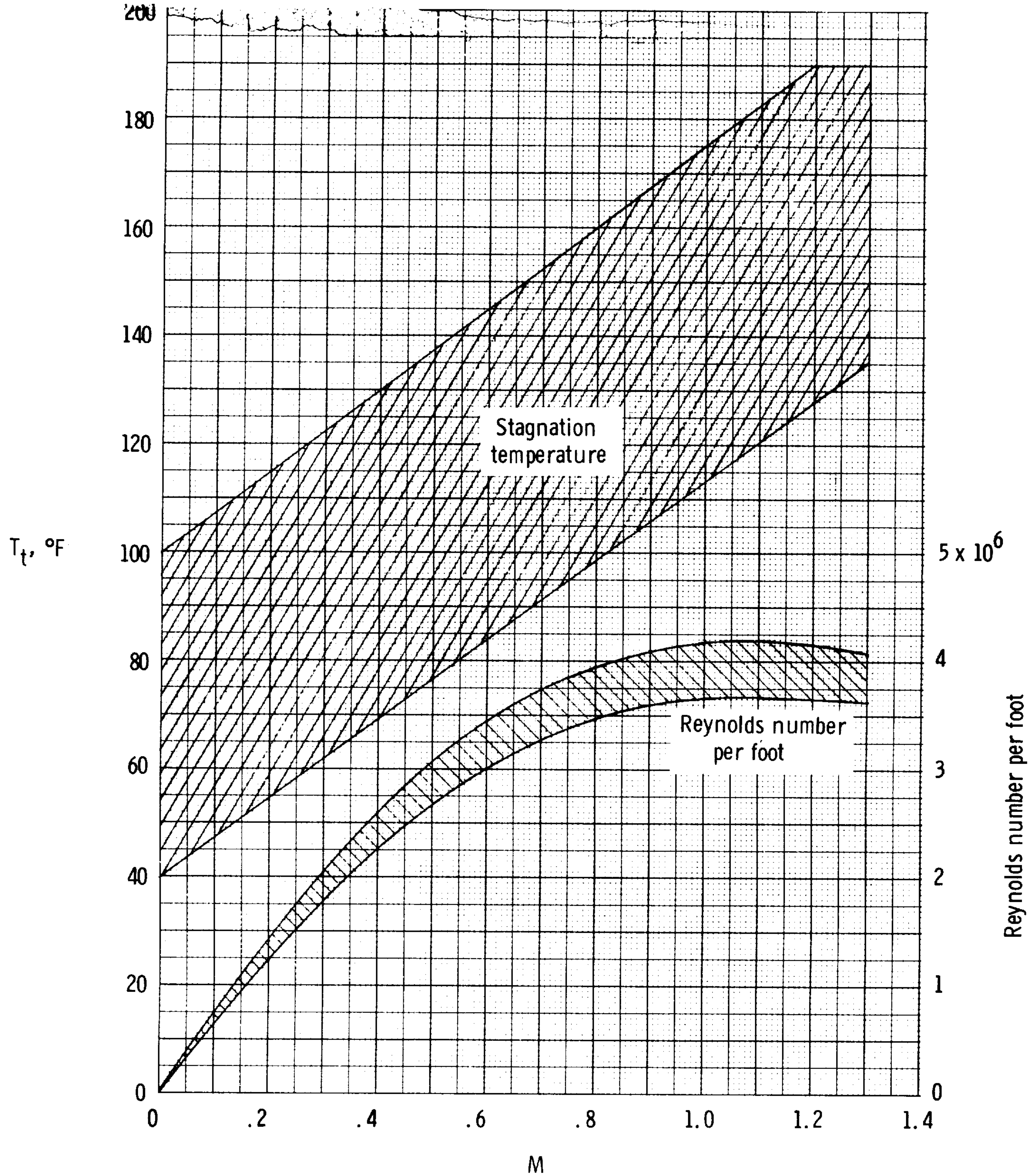


Figure I-8. Variation of Reynolds number per foot and stagnation temperature with Mach number.

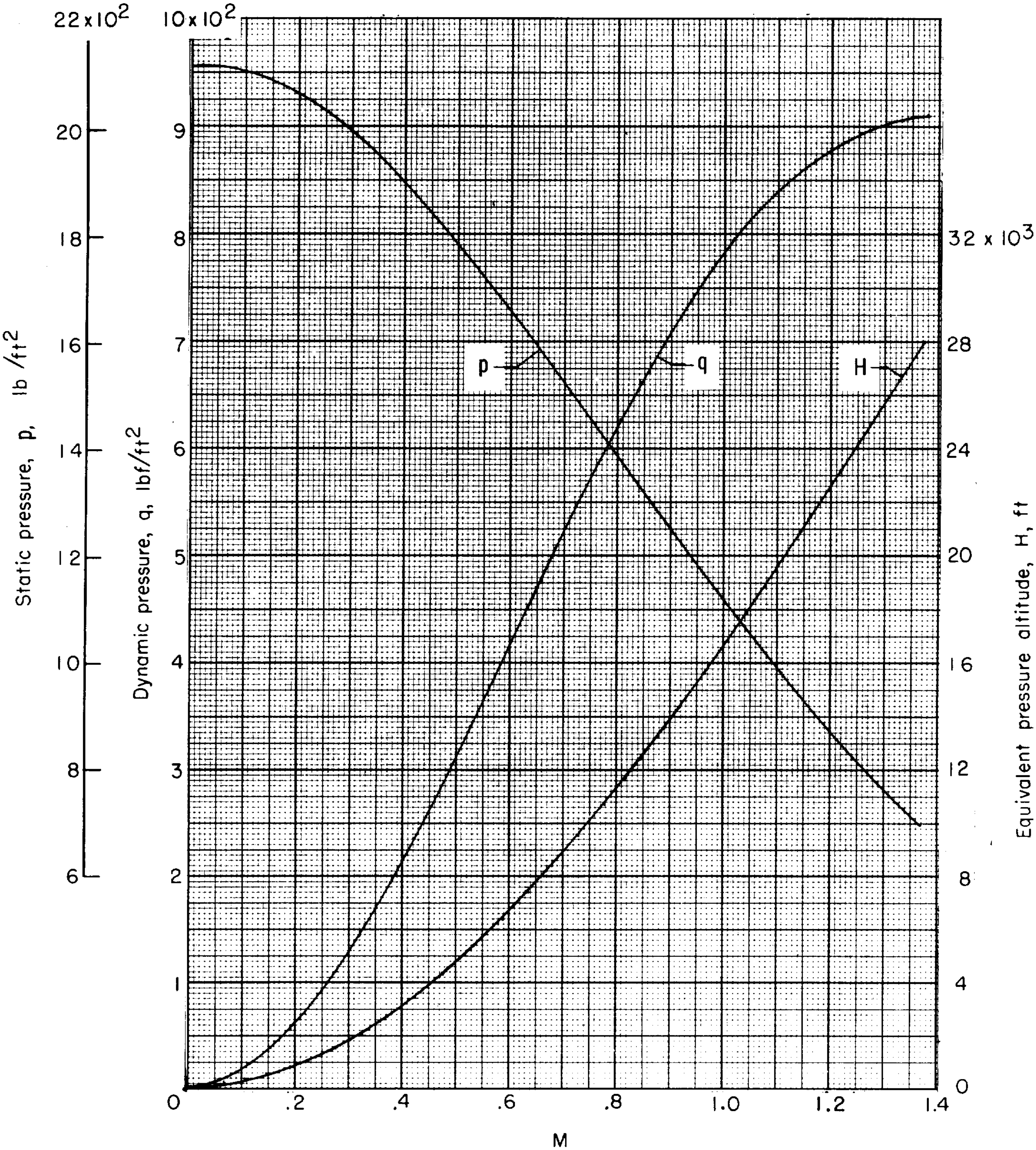


Figure I-9. Variation of static pressure, dynamic pressure, and equivalent pressure with Mach number, in the test section of the Langley 16-Foot Transonic Tunnel.

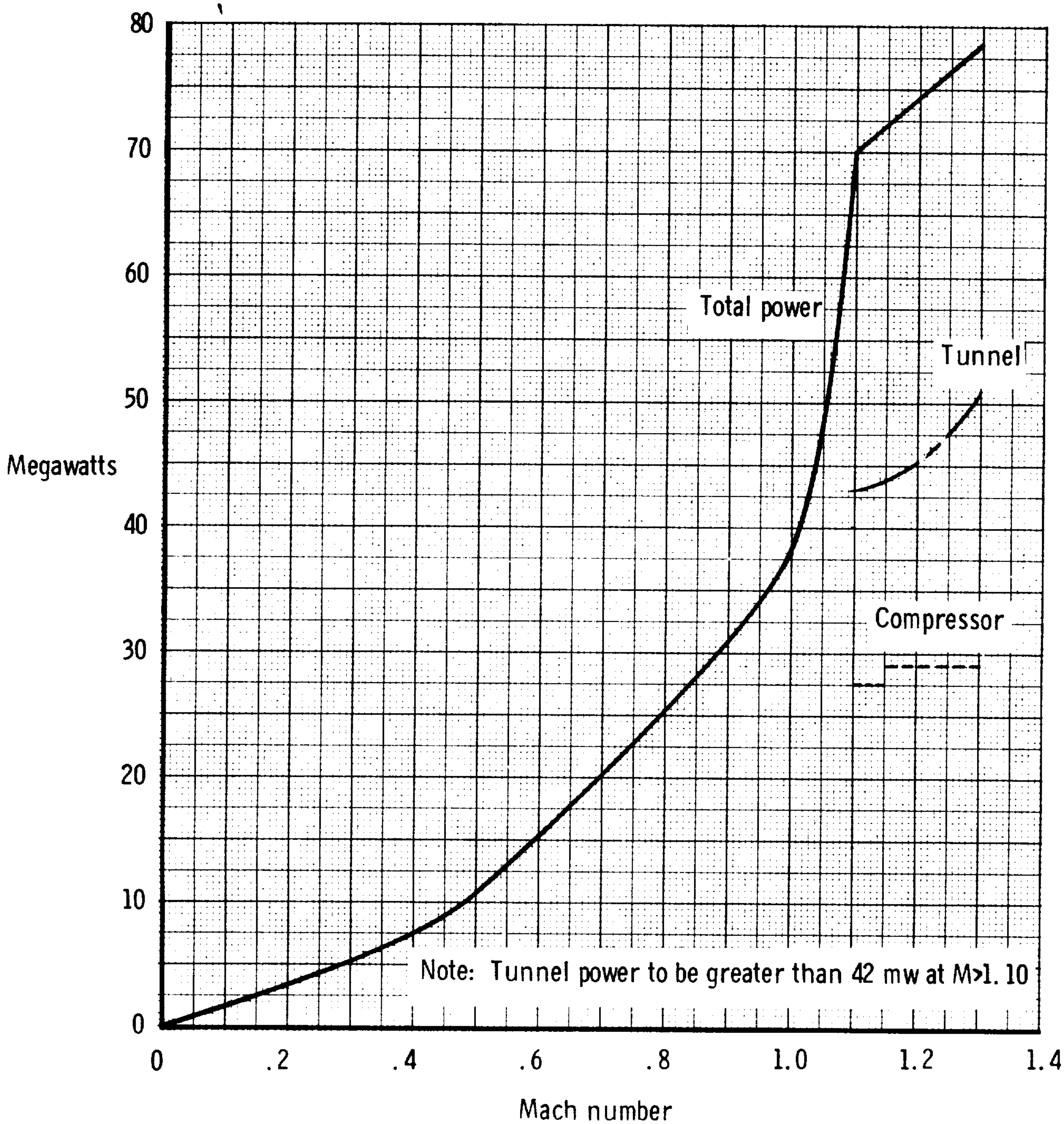


Figure I-10. Tunnel power.

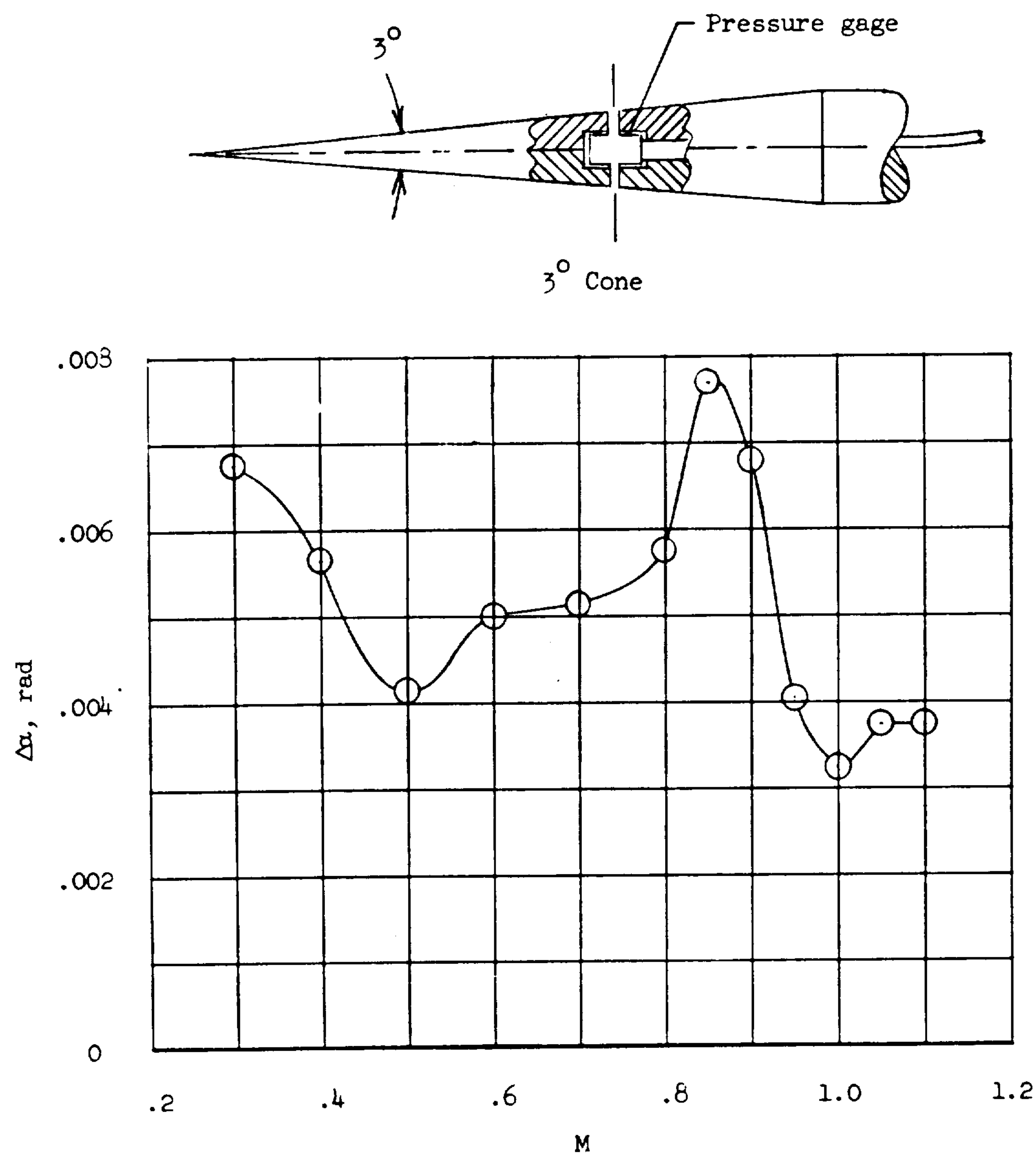


Figure I-11. Variation with Mach number of turbulent-fluctuating-airstream angularity on wind-tunnel center line in pitch plane of model. (From 3° cone data).

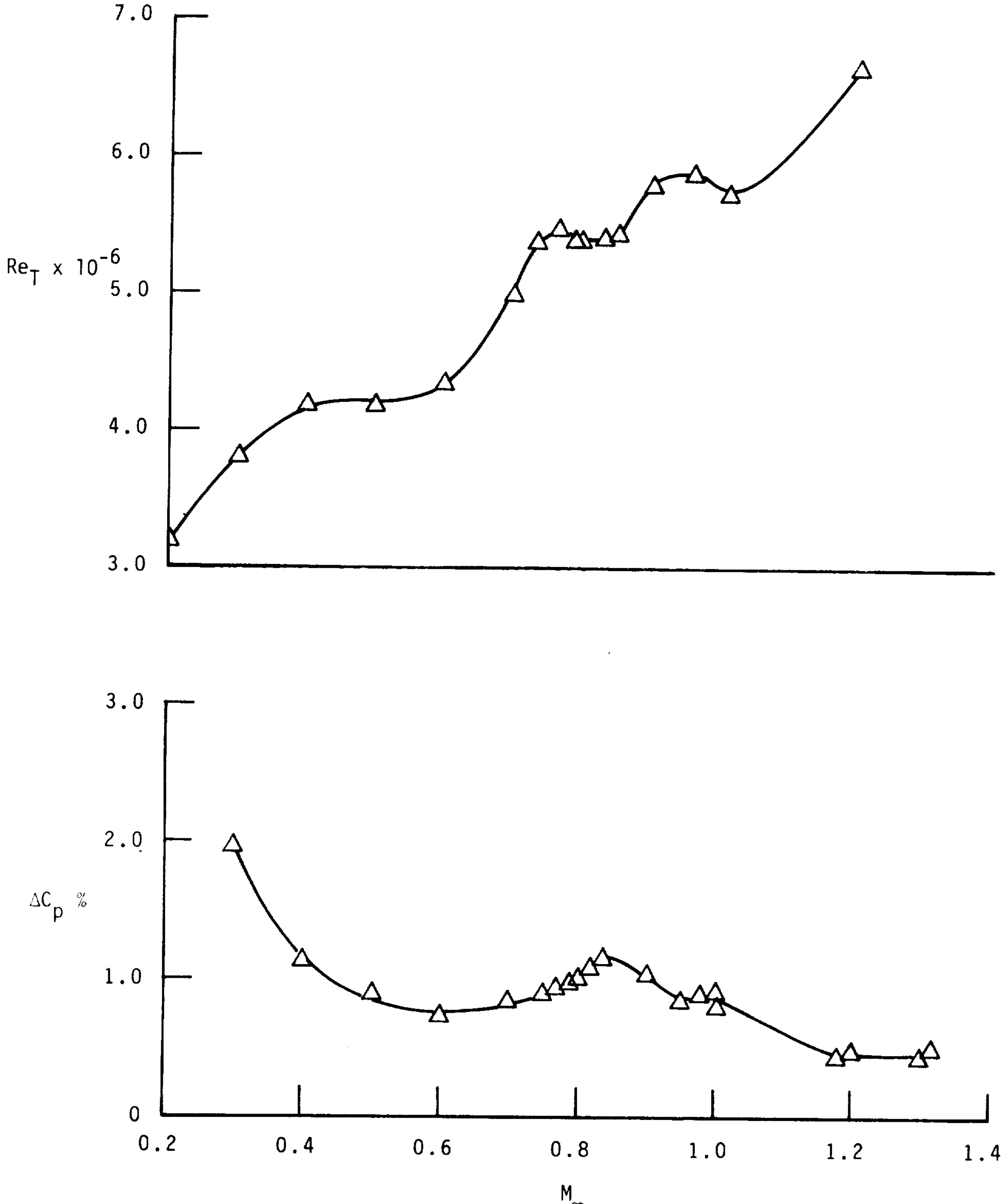


Figure I-12. Noise and Re_T variation in NASA/LRC 16-Foot Transonic Tunnel.

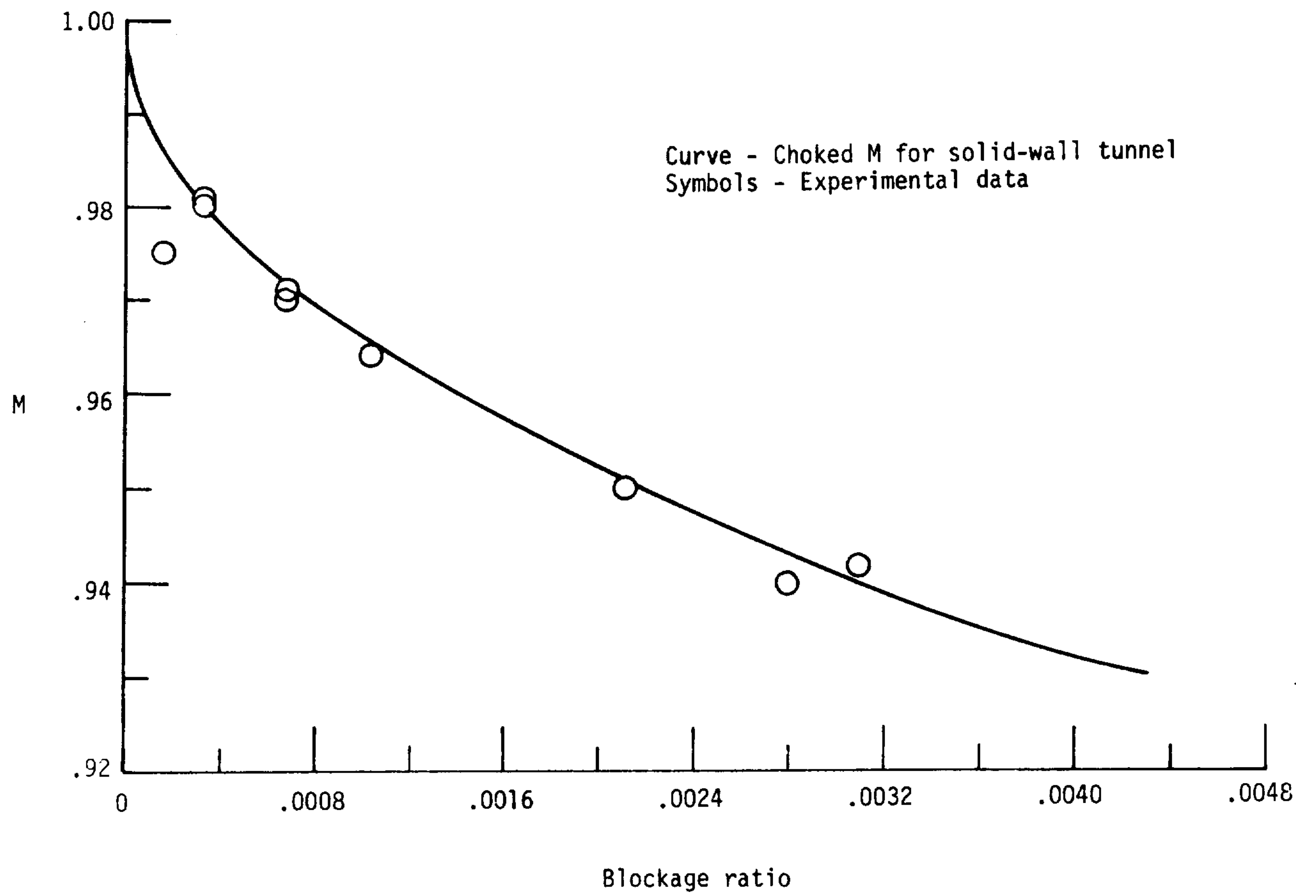


Figure I-13. Variation of transonic-creep Mach number with blockage ratio.

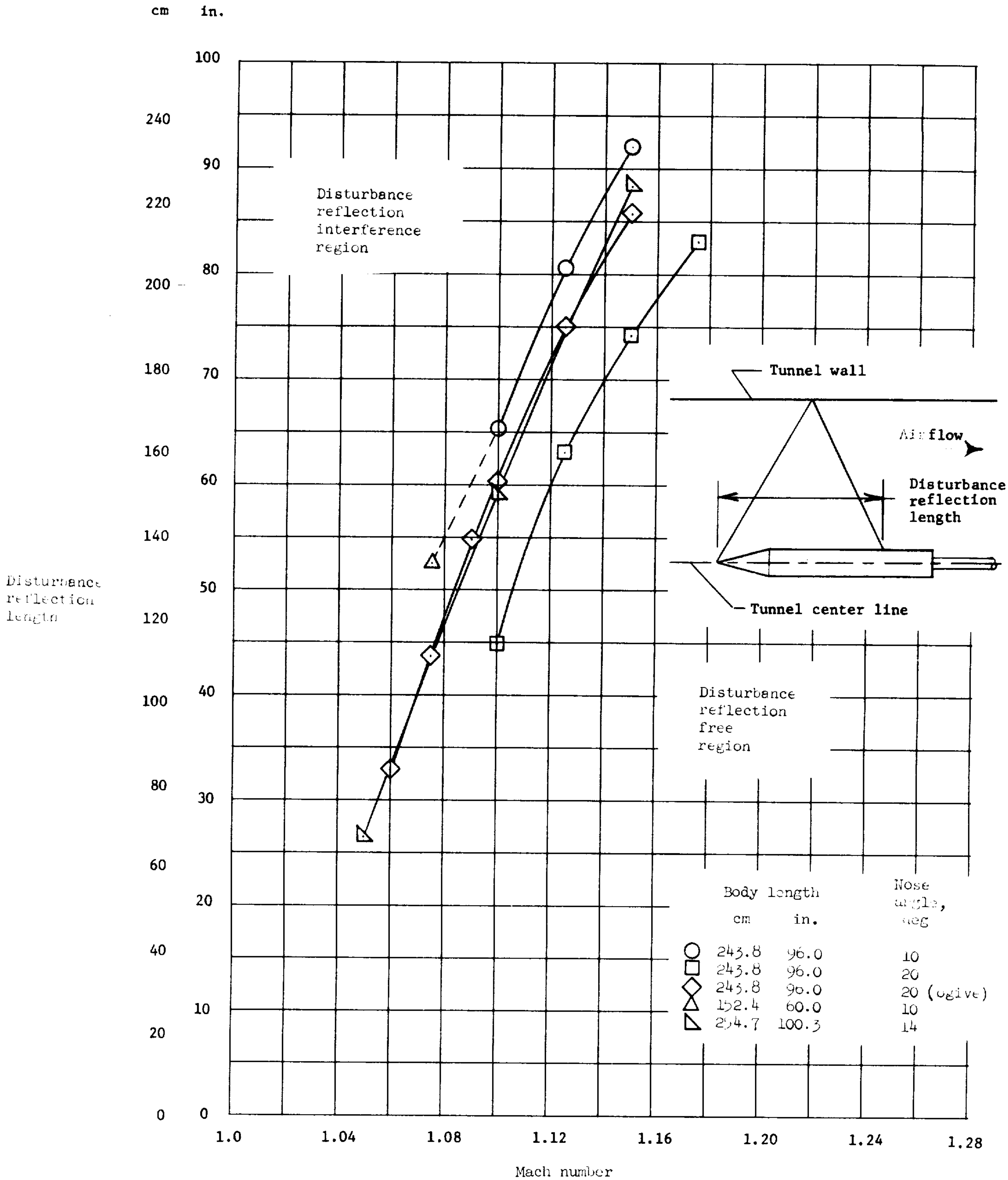


Figure I-14. Boundary-reflected-disturbance lengths measured in the Langley 16-Foot Transonic Tunnel.

SECTION II - Model Installation Equipment for Aerodynamic Testing

A significant amount of hardware exists appropriate to aerodynamic force testing where jet exhaust simulations are not required. This hardware has accumulated over the years and generally emerges from specific test programs having specific objectives. Therefore, there is a wide variation in the versatility or usefulness of some components. Hardware designed for propulsion simulation testing can also be used for force-only (Aerodynamic) investigations whenever feasible.

A. Aerodynamic Testing On or Near the Test Section Center line.- Briefly, for force-only testing as with propulsion testing, it is desirable to have the model base forward of Tunnel Station 137 (to avoid diffuser Mach gradient) and the model center close to Tunnel Station 134 (to keep the model near the tunnel center line through the angle-of-attack range). Consultation with Propulsion Aerodynamic Branch personnel is necessary to ascertain the appropriateness of a projected model location in the test section for the range of freestream flow conditions in the test program (see Section I-F). A sketch showing the support system and pertinent test section stations is presented in figure II-1. The load capacity of the main support system is 10,000 lbs. of normal force and 10,000 lbs. of side force acting at Tunnel Station 134.0. The available strut head angular travel allows straight rear sting type model installations (no bent coupling) to traverse an angle-of-attack range from -10° to +25°. The strut head has remote rotary control so that the model and sting can be rolled as a unit from -90° to +90°.

Lateral aerodynamic force testing can be done by rolling the model in the -90° to +90° range and traversing the main tunnel support system in the vertical plane. Manually installed fixed bent couplings can be used to give a model-sting combination an angular offset relative to the center line of the main support system. Models are supported in the test section by combining pieces of hardware shown in figures II-2 to II-4. The following sections will describe these in more detail.

B. Attachment to Main Support System (Strut Head).- Attachment to the main support system at the strut head (Tunnel Station 141.94) is accomplished using one of the hardware components shown in figure II-2 or more specialized propulsion simulation hardware (shown in Section III). Each of the components shown in figure II-2 has the same bolt pattern on the downstream end and fits interchangeably to the face of the strut head. The component identified as the strut head extension (figure II-6) has this bolt pattern on each end and serves to move the strut head bolt pattern 10.4 in. forward so that a model installation can be moved forward in the test section. The standard and air system sting butts (figure II-5) are the basic attachment components to support model installations in the 16-Foot Transonic Tunnel. Butts 1 and 2 are generally used for aerodynamic force testing and have knuckle tang seats in two planes. Butts 3 and 4 have a 2.5 in. bore to allow a high-pressure air line to pass through and have knuckle tang seats in only one plane. The 2.5 in. bore in these two butts has been opened up on the top and bottom to allow passage of instrumentation with the air line installed. The offset butt (in design) shown in figure II-7 displaces the attachment for the downstream end of the 16-Ft. T.T. knuckle 22.0 inches below the tunnel center line so that models mounted on sting struts (fig. III-7) will be on the test section center line.

A requirement to install an existing model-balance-sting combination (C-5A) for a tunnel-to-tunnel correlation program resulted in the fabrication of a butt (figure II-8) that directly adapted a sting (having

Ames 11- by 11-ft. and AEDC tunnel sting taper) to the main support system. The two components identified as sting cones (long and short) are built up from mild steel and steel plate and, therefore, have limited load carrying capability (figures II-9 and II-10). Another adapter butt (figure II-11) was fabricated to allow installation of an F-18 propulsion simulation model. This butt accepts the same sting taper described in figure II-8. The angle of the center line of the taper in this adapter butt is 20° relative to the model support system center line.

C. Attachment to Standard Butts.- Attachment to a standard butt (figure II-5) can be accomplished through a male taper shown as the downstream portion of the split coupling (knuckle) of figure II-13, the downstream end of an adapter (some of Type 3, figure II-3), or the end of a sting having the same male taper as the downstream end of a knuckle. The knuckles are split in half longitudinally so that they can be changed manually between tunnel runs without disconnecting the instrumentation passing through them; however, this is a time-consuming task due to the size of the hardware involved. An assembly cross-section of a typical sting-knuckle-butt combination is shown in figure II-12. Tapers at both the sting and butt ends of the knuckles are seated using the split nuts shown in figure II-14. At this point, the installation possibilities expand and various combinations of knuckles and/or adapters allow the use of hardware designed for other tunnels. The listing given in figure II-3 summarizes the adapter hardware that is under the control of the 16-Foot Transonic Tunnel and individual sketches are presented in figures II-15 through II-26. Fabrication of new components to fit standard butts and knuckles is aided by use of taper gages for both ends of the split knuckle and thread gages for the split nuts that seat the upstream and downstream knuckle tapers to stings/adapters and butts.

D. Available Stings.- Stings under the direct control of the 16-Foot Transonic Tunnel are listed in figure II-4 with a few pertinent parameters as a general description. These stings were built at various times for specific model installations (see column in figure II-4 entitled Program) and as such have no systematic relationship in construction or utility. However, they are available for use and those that do not accept an existing Langley force balance at their upstream end can be made useable by modifying them or having adapters fabricated. Those stings that have immediate utility with an existing Langley force balance are detailed individually in figures II-27 to II-36.

Sting installation is not limited to those listed since the adapters listed in figure II-3 make it possible to install some stings built for (and under the control of) other wind tunnels at Langley or about the country. It must be kept in mind that tandem installation of a series of knuckles and adapters increases the length of the assembled support system and can, in some cases, put the model too far forward in the test section. The length of test section having a constant centerline Mach number is limited especially at high Mach numbers so that model longitudinal position in the tunnel must be considered. In addition, if the model is to be kept near the tunnel center line through an angle-of-attack range, the center of the model should be near Tunnel Station 134 which is the center of rotation of the support system sector (figure II-1). Tunnel Station 141.94 is the location of the strut head face and cannot be moved downstream. Taper and thread gages matching each Langley force balance or family of force balances are available for the fabrication of the upstream ends of stings. All sting and model hardware designs must satisfy the handbook "Wind-Tunnel Model Systems Criteria" (LHB.1710.15) and be approved by Langley personnel before they can be used in the wind tunnel. Borrowed hardware must also satisfy the same Langley standards and be inspected (e.g. magnafluxed) at Langley. See Section IX for procedures for planning a test.

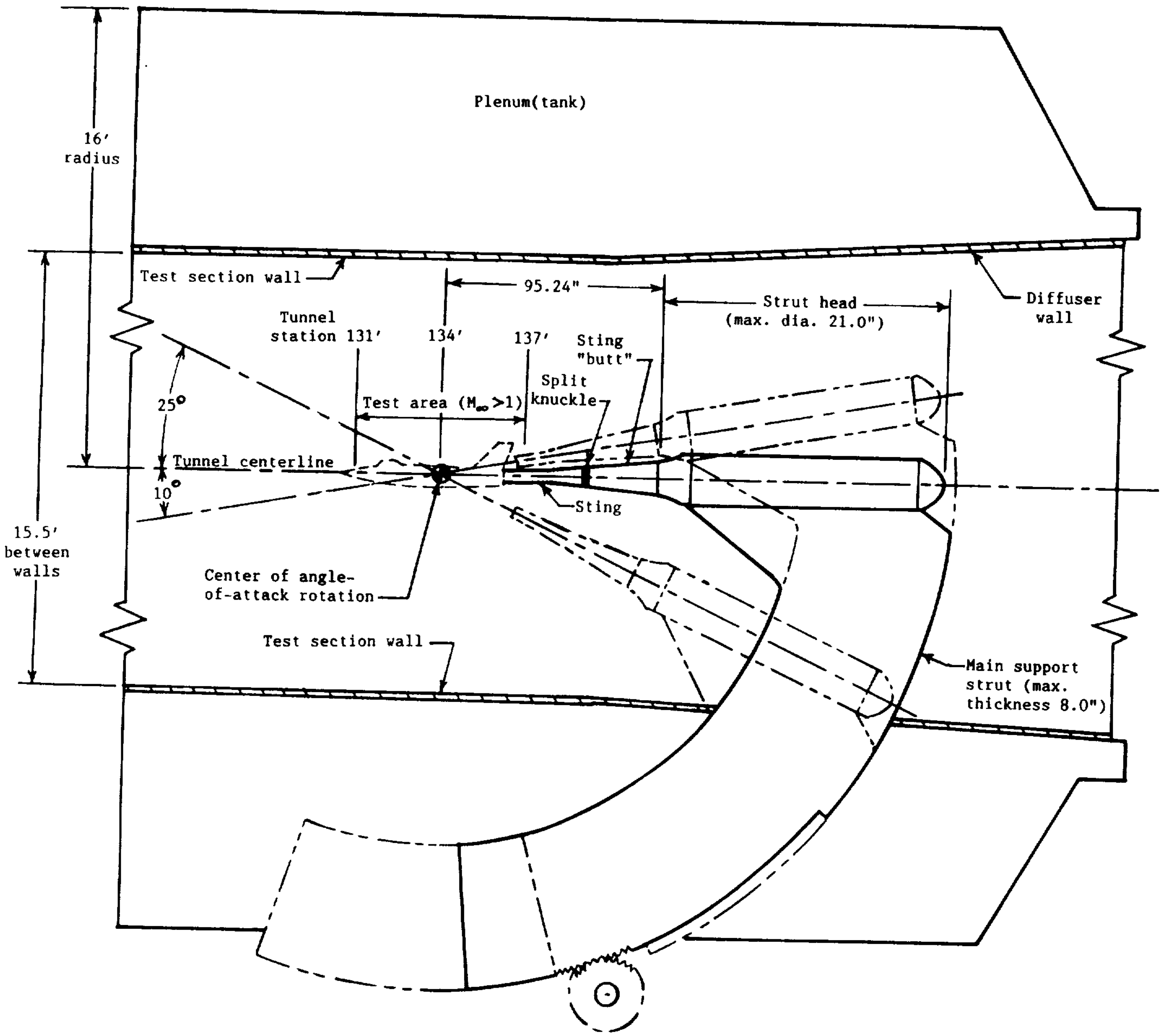
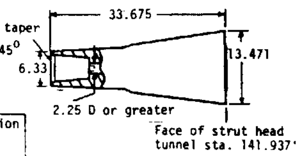


Figure II-1. Schematic of model support system of 16-Foot Transonic Tunnel.

Standard Sting Butt - LD-44571-5 (Dovel pin locations at aft end incorrect on drawing)

Used for force tests or any other test except those utilizing high pressure air for propulsion simulation

Material: 2340
RC 33
Yield 120,000 psi
Knuckle angles: 0°, 2°, 5°, 10°, 12.5°, 15°, 20°, 30°, 45°



Air System Sting Butt - LE - 528489

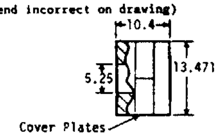
For propulsion simulation tests utilizing high pressure air

Material: 4340
RC 37
Yield 145,000 psi
Knuckle angles: 0°
Hole in butt 2.50"D instead of 2.25"D of standard

Strut Head Extension - LDZ-50353-6 (Dovel pin locations at each end incorrect on drawing)

Material: (probably) 2340

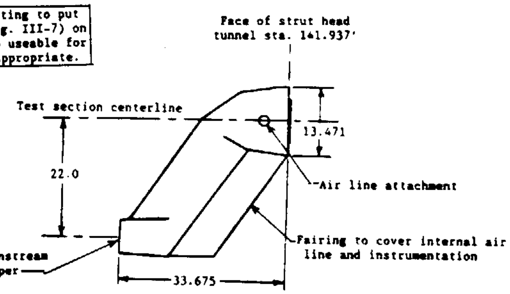
This hardware, when placed between any of the butts on this page and the strut head face, moves the model installation 10.4" forward in the test section.



Offset Sting Butt - (In design phase)

Used for propulsion simulation testing to put models mounted on sting-struts (fig. III-7) on the test section centerline. Also useable for routine aerodynamic testing when appropriate.

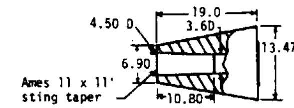
(Capable of dual high pressure air flow. One air line housed internally and second air line exiting side of butt.)



Adapter Butt To Ames 11 x 11' Stings - LD-904295

Material: 4340
RC 36-38
Yield 145,000 psi

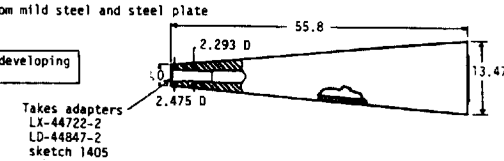
This hardware accepts the male taper of Ames 11 x 11' Transonic Tunnel stings



Long Sting Cone - LX-44722-1 (Bolt and pin hole patterns in aft end inaccurate and incomplete)

Material: Built up from mild steel and steel plate
Yield 49,000 psi

Not used for models developing significant loads

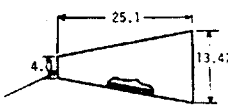


Short Sting Cone - LD-44473-8 (Bolt and pin hole patterns in aft end inaccurate and incomplete)

Material: Built up from mild steel and steel plate
Yield 49,000 psi

Not used for models developing significant loads

(See front end details above)
Takes adapters
LX-44722-2
LD-44847-2
sketch 1405



Adapter Butt To Ames or ARO Taper - Northrop Drawing 5230420 SH.5

Material: 4340
Heat Treat 160-180,000 psi

This hardware accepts air sting 0951-020

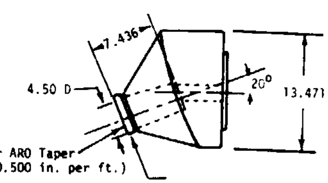
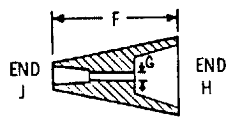
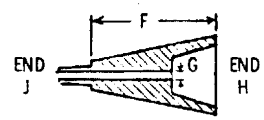


Figure II-2. Attachments to 16-Foot Transonic Tunnel strut head. (All dimensions are in inches unless otherwise indicated.)

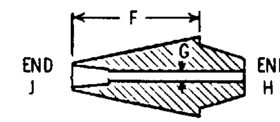
TYPE 1



TYPE 2



TYPE 3



STING ADAPTERS

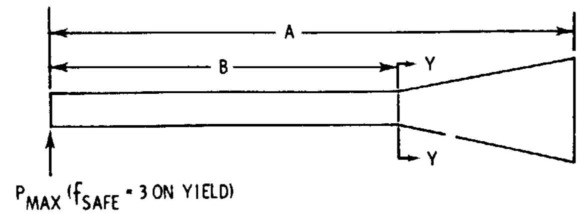
All dimensions are inches unless otherwise indicated.

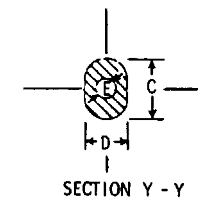
Drawing No.	Function	Type	End H Fits	End J Fits	F	G	Material	Design Heat Treat	Yield (PSI)	Built For	Remarks
LD-907034	16'TT to 8'TPT Standard	1	16' Knuckle	8' Knuckle	11.33	1.060.	4340	RC 36-40	140,000	C-5A Force Model	
LD-517081 A	16'TT to 8'TPT Bastard	2	16' Knuckle	8' Bas. Sting	10.50	2.00.	17-4	RC 39	155,000	F-8 SCW Model	
303M0D7024	16'TT to Cornell 8'	3	16' Butt	Cornell Sting	4.042	2.0D.	4340	RC 27-32	125,000	F-14 Force 1/16-Scale	CALSPAN S_1 Taper→ 5° Bend→ No Knuckle Can Be Used
68T071540	16'TT to McD. Polysonic	1	16' Knuckle	McD. Sting	25.710	1.0D.	17-4	RC 41	180,000	F-15 Force Model	
12-ZS-116-6	16'TT to Ames 6x6'	1	16' Knuckle	6x6' Sting	20.82	1.0D.	4340	RC 39	150,000	F-111 Force 1/20-Scale	
WTC 439-139	16'TT to Boeing TWT	2	16' Knuckle	Boeing Sting	11.5	2.0D.	4340	RC 39	160,000	737 Force Model	Assembly Drwg. No. 1211-294&1211-302
LC-516838	8'TPT Bastard to 7x10'	2	LD-517081	7x10 HST Sting	4.7	1.25D.	4340	RC 33-35	135,000		Also Accepts 8FT TPT Increased Strength Stings
LD-534208	16'TT to 16'TT	3	16' Butt	16' Knuckle	10.913	2.375 D.	4340	RC 39-42	180,000		12.5° Bend. Combines With 16'TT Knuckles
LD-538484	16'TT to 16'TT	3	16' Butt	16' Knuckle	10.913	2.375 D.	4340	RC 39-42	180,000		5° Bend. Combines With 16'TT Knuckles
Sketch 1405	16'TT to 1.5" Cylindrical	3	16' Sting Cone	1.5" D. Cyl.	31.15	1.0D.	4130 or 4340	-----	52,000		1.5" D. Cyl. Fit by 2 Setscrews at End J
LD-44847-2	16'TT to 8'TPT St. Knuckle	3	16' Sting Cone	8' Knuckle	5.5	0.95D.	4130	-----	52,000	Model 75 Sting Shaft	
LX-44722-2	16'TT to 1.75" Cylindrical	3	16' Sting Cone	1.75" D. Cyl.	26.95	1.0D.	Mild Steel	-----	49,000	Model 75-1	1.75" D. Cyl. Fit Setscrew→ Orig. Drwg. Does Not Show Modifications

BALANCE ADAPTER

Drawing No.	Function	Type	End H Fits	End J Fits	F	G	Material	Design Heat Treat	Yield (PSI)	Built For	Remarks
LD-533596	Task Mk V or VI to 1617 Bal	3	Task Mk V or VI Stings	1617 Bal	2.21	1.187 D.	17-4	H-1075	112,000	B-1 Force 0.06-Scale	Fits Stings With 0.6 In. Per Ft. Task Bal Taper

Figure II-3. Available sting and balance adapters. (All dimensions are in inches unless otherwise indicated.)





BENDING (f_{SAFE}) = 3

$$P_{MAX} = \frac{(\text{YIELD})\, I}{3\, B \left[\frac{C}{2}\right]}$$

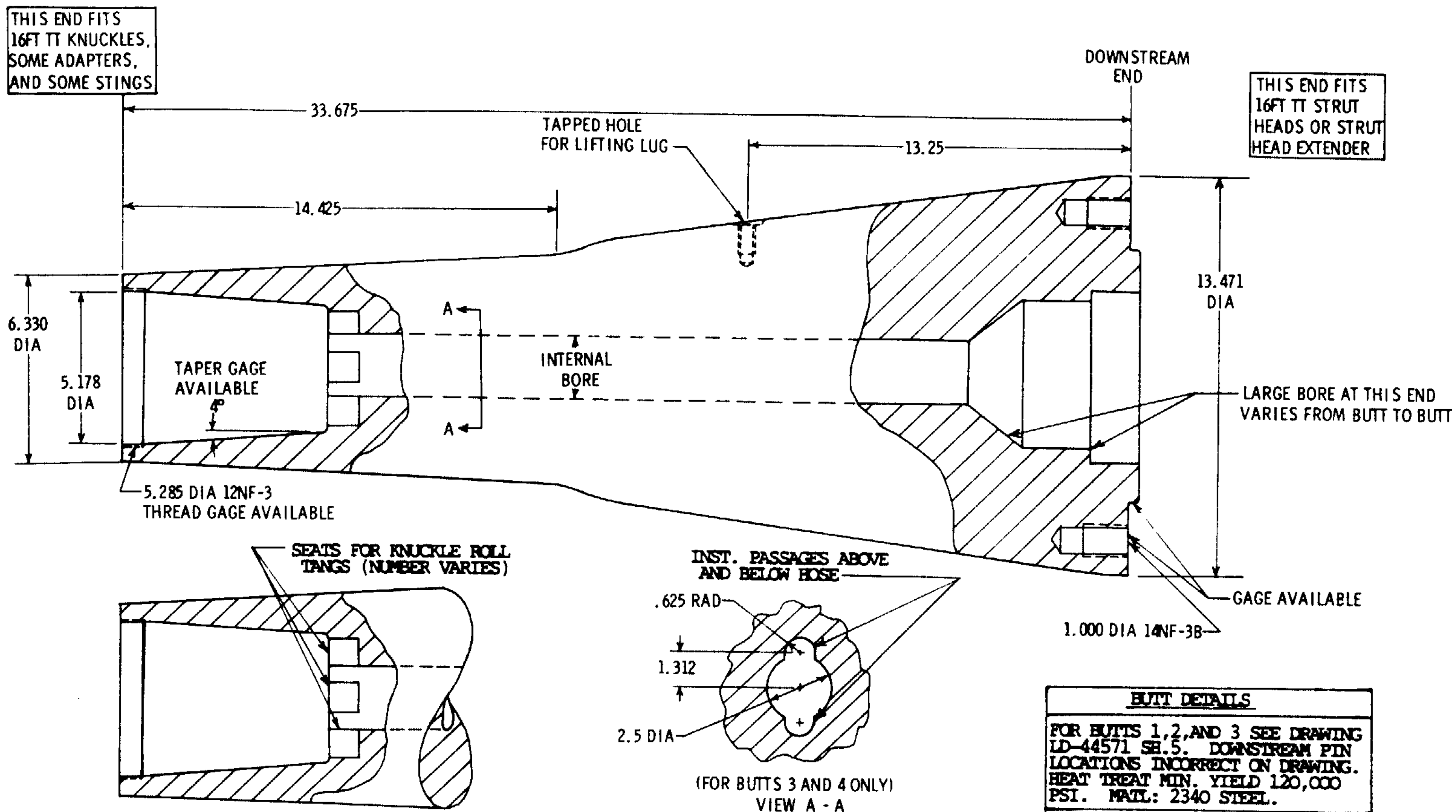
* (Brackets indicate no working balance available at LRC)

** (P_{Max} limited to 10,000# at tunnel station 134)

All dimensions are inches unless otherwise indicated.

Drawing No.	Program	A	B	C	D	E	Material	Design Heat Treat	Yield (PSI)	Section Y-Y: I, in.^{4}	Section Y-Y: P^{**}_{max}, lbs.	Front End Fits *	Back End Fits	Remarks
LE-904416	LASAM	54.0	43.0	2.0	2.0	0.875 D.	17-4	RC 40	170,000	1.029	1350	731 Bal.	8' TPT St. Knuckle	Max. D. Bal. Taper 1.416"
LD-518426	ATT INLET	55.86	42.86	2.5 D.	---	1.0 D.	4340	RC 40	165,000	1.868	1900	731 Bal.	16' Knuckle	Max. D. Bal. Taper 1.416"
LE-403234	Swallow	54.5	41.35	3.125	2.0	0.625 D.	4340	RC 34-36	140,000	2.996	2160	( )	16' Knuckle	Male Taper Front End → Max. D. 2.188"
LE-404030	Crank Wing	51.61	31.442	3.6 D.	---	1.0 D.	4340	RC 35-37	145,000	8.196	7000	1617 Bal.	16' Knuckle	Max. D. Bal. Taper 2.60" → Dogleg 1.25" ℄ Offset
LDZ-50334	Unified Wing	51.052	32.302	4.311 D.	---	2.0 D.	4340	RC 35	142,000	16.169	10,990	( )	16' Knuckle	Front End Forked
LX-904027-1	AR-8	51.484	31.905	4.0	2.7	2.0 D.	4340	RC 36	145,000	7.697	5830	1617 Bal.	16' Knuckle	Max. D. Bal. Taper 2.60" → Taper ℄ Offset 0.30"
LD-44571-2	Model 75-3	49.5	30.75	4.31 D.	---	2.0 D.	2340	RC 33	120,000	16.169	9760	( )	16' Knuckle	Max. D. Bal. Taper 3.686" (4° Taper)
LDZ-50023	Wing C	49.0	30.25	3.5 D.	---	1.0 D.	4340	RC 39	160,000	7.317	7370	( )	16' Knuckle	Max. D. Bal. Taper 3.686" (4° Taper)
LDZ-50023	Wing C	49.0	30.25	2.25 D.	---	1.0 D.	4340	RC 39	160,000	1.209	1895	( )	16' Knuckle	Max. D. Bal. Taper 3.686" (4° Taper)
LEZ-50580-3	S-52½ C	61.352	47.321	4.50	3.136	2.0 D.	17-4	RC 38	170,000	13.242	7050	( )	16' Knuckle	Front End Forked
LDZ-5589-1(MOD)	YF-100A	49.0	25.75	3.375 D	---	2.0 D.	4150	RC 39	163,000	5.584	6980	( )	16' Knuckle	Max.D Bal. Taper 3.686"→(4°Taper)Dogleg 1.678" ℄ offset
LEZ-50329-11	S-45 C	61.352	26.736	4.5	2.546	2.0 D.	4340	RC 40	157,000	10.063	9225	( )	16' Knuckle	Front End Forked
LDZ-5984-1	Convair MX-1364	56.5	22.0	2.25 D.	---	0.5 D.	4340	RC 38-40	160,000	1.255	2700	( )	16' Knuckle	Plate At Front End
LDZ-50166	P6M(SK42543)	91.0	60.5	2.5 D.	---	1.25 D.	Cold rolled steel	RC 32	85,000	1.798	670	UT-03	16' Knuckle	Sleeved to Max D Bal Taper 1.062"→39.5" Cold Roll Added
4Z1641	Convair	54.11	25.5	2.25 D.	---	0.5 D.	4340	RC 40	165,000	1.255	2405	( )	16' Knuckle	Plate At Front End
12ZS116-13	GUF-111(1/20)	39.01	13.03	1.5	.70	0.38 D.	4340	RC 42	175,000	.196	1170	(Task 1.5" Bal)	Adapter 12-ZS-116-6	Max. D. Bal. Taper 1.05" → Dogleg 0.92" ℄ Offset
LD-515641A	SCW Model	61.1	41.0	3.25	2.0	1.25 D.	4340	RC 38-40	160,000	4.517	3615	731 Bal.	Adapter LD-517081	Max. D. Bal. Taper 1.416"
LD-525335	Pylon MTD NAC	34.875	12.0	2.4 D	- - -	1.0 D.	17-4	RC 38	170,000	- - - -	1457	731 Bal.	Adapter LD-907034	Male Back End is Critical Stress Location Alteration to Male End LC-90-324
LDZ-5589 MOD. 19DEC77	Space Shuttle	34.50	11.25	4.311 D	- - -	2.0 D.	4150	RC 39	163,000	16.169	36,000	(TASK Mk VI)	16' Knuckle	With Adapter LD-533596 Takes 1617 Bal.
75-002760	ADAM-II	52.858	52.858	4.25	- - -	2.0 D.	4340	RC 43-46	200,000	15.23	9030	VTB-3 Bal.	16' Butt	Max. D. Bal. Taper 3.375". For Propul. Test
LDZ-5622	RF-84F	51.053	27.303	4.0 D	- - -	2.0 D.	4340	RC 40	160,000	8.9	9290	( )	16' Knuckle	Front End Forked. Dia Undercut 1/4" On Top
LD-906917	SST #8	51.0	46.0	2.625 D	- - -	0.5 D.	17-4	RC 36-40	165,000	2.328	2120	( )	8' TPT St. Knuckle	Front End MOD. LD-510382. Male Taper → Max D 2.188"
LD-925157	Jet Nac Cal Sting	35.0	30.72	1.5 D	- - -	0.75 D.	17-4	H-900		.233		UT-03	UPWT B-2 Adapter	Max. D. Bal. Taper 1.062". B-2 Adapter Fits Adapter LD-907034
LC-904302	Cowl Blockage	38.880	19.880	.875 D	- - -	0.5 D.	4340	RC 39	160,000	.0257	150	1622	UPWT B-2 Adapter	Max. D. Bal. Taper 0.5". B-2 Adapter Fits Adapter LD-907034
1128MOD2504	EA-6B	73.682	37.152	4.63	2.250	1.25D.	15-5	H-900	170,000	15.415	10,150	NTF-101B	16' Knuckle	Max. D. Bal. Taper 2.0"

Figure II-4. Stings at 16-Foot Transonic Tunnel. (All dimensions are in inches unless otherwise indicated.)



BUTT	INTERNAL BORE (DIA, IN)	TANG SEATS	INST. INTERNAL WITH HIGH PRES. AIR HOSE
1	2.25	4	CAN'T TAKE HOSE
2	2.50	4	NO
3	VIEW A-A	2	YES
4	VIEW A-A	2	YES

BUTT DETAILS
FOR BUTTS 1, 2, AND 3 SEE DRAWING LD-44571 SH.5. DOWNSTREAM PIN LOCATIONS INCORRECT ON DRAWING. HEAT TREAT MIN. YIELD 120,000 PSI. MATL: 2340 STEEL.
FOR BUTT 4 SEE DRAWING LE-528489. HEAT TREAT YIELD 150,000 PSI. MATL: 4340 STEEL.

Figure II-5. Sketch of standard butts. (All dimensions are in inches unless otherwise indicated.)

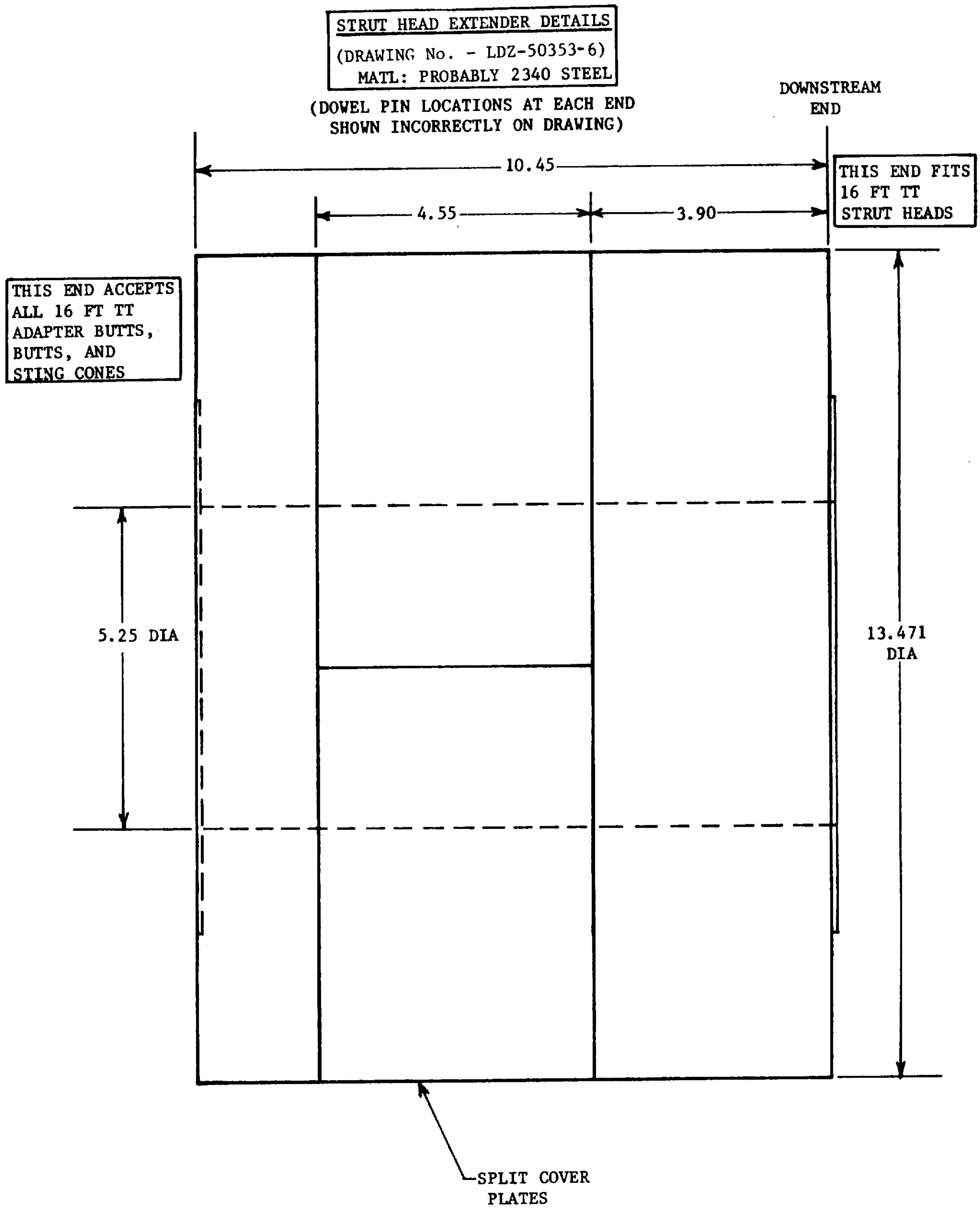


Figure II-6. Sketch of strut head extender. (All dimensions are in inches unless otherwise indicated.)

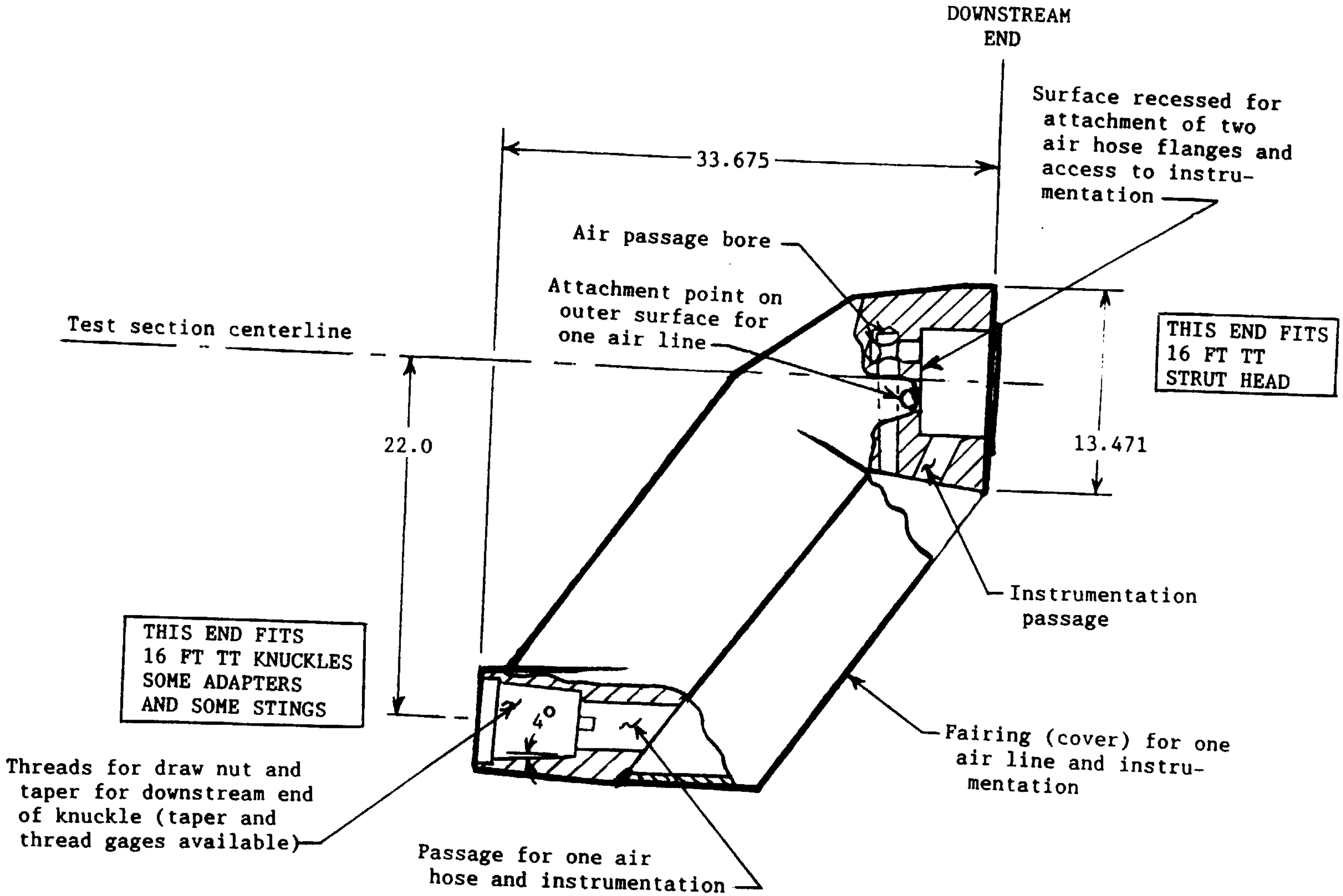


Figure II-7. Preliminary sketch of offset butt (in design) for force and propulsion simulation testing. (All dimensions are in inches unless otherwise indicated.)

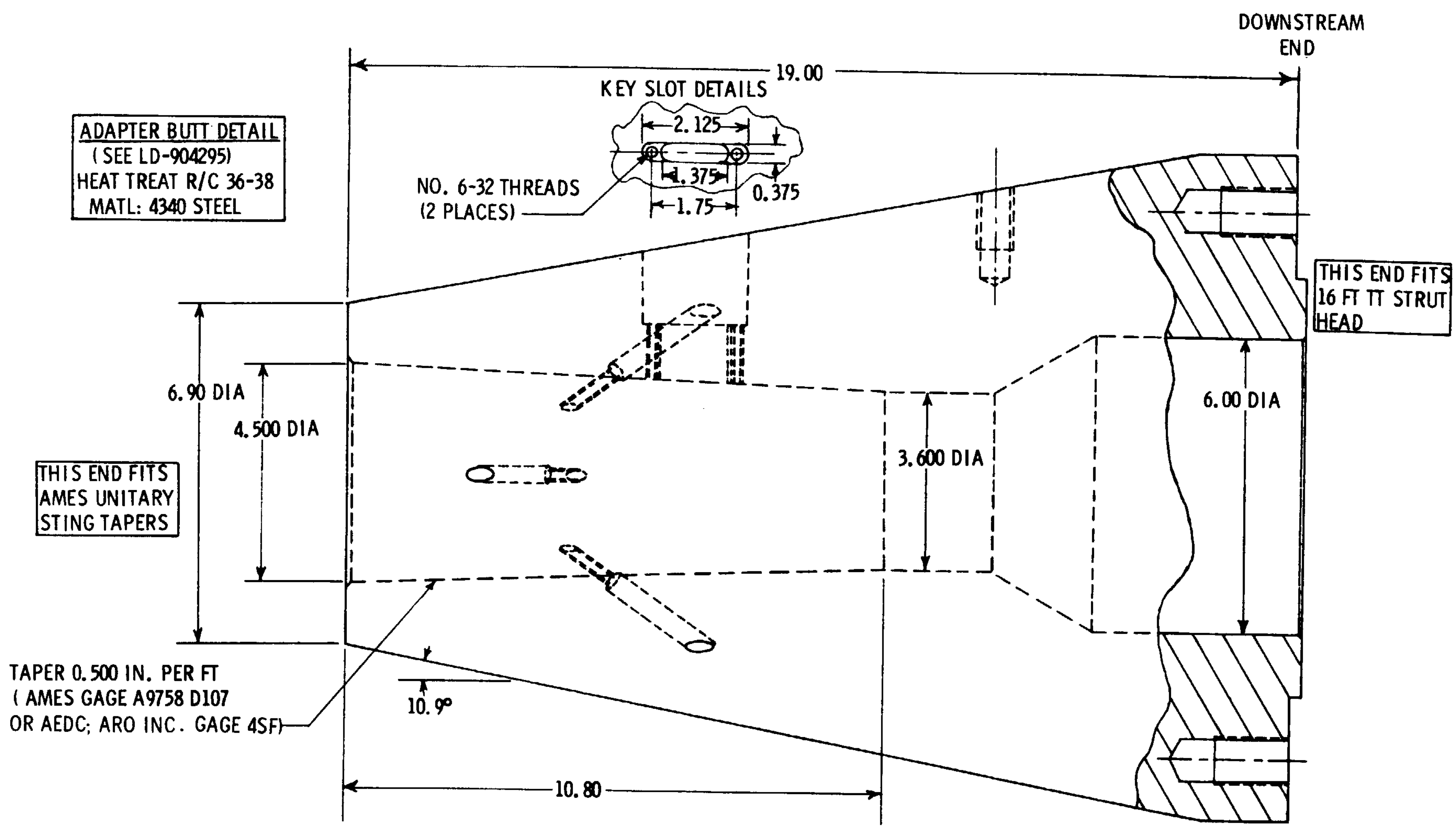


Figure II-8. Sketch of adapter butt LD-904295. (All dimensions are in inches unless otherwise indicated.)

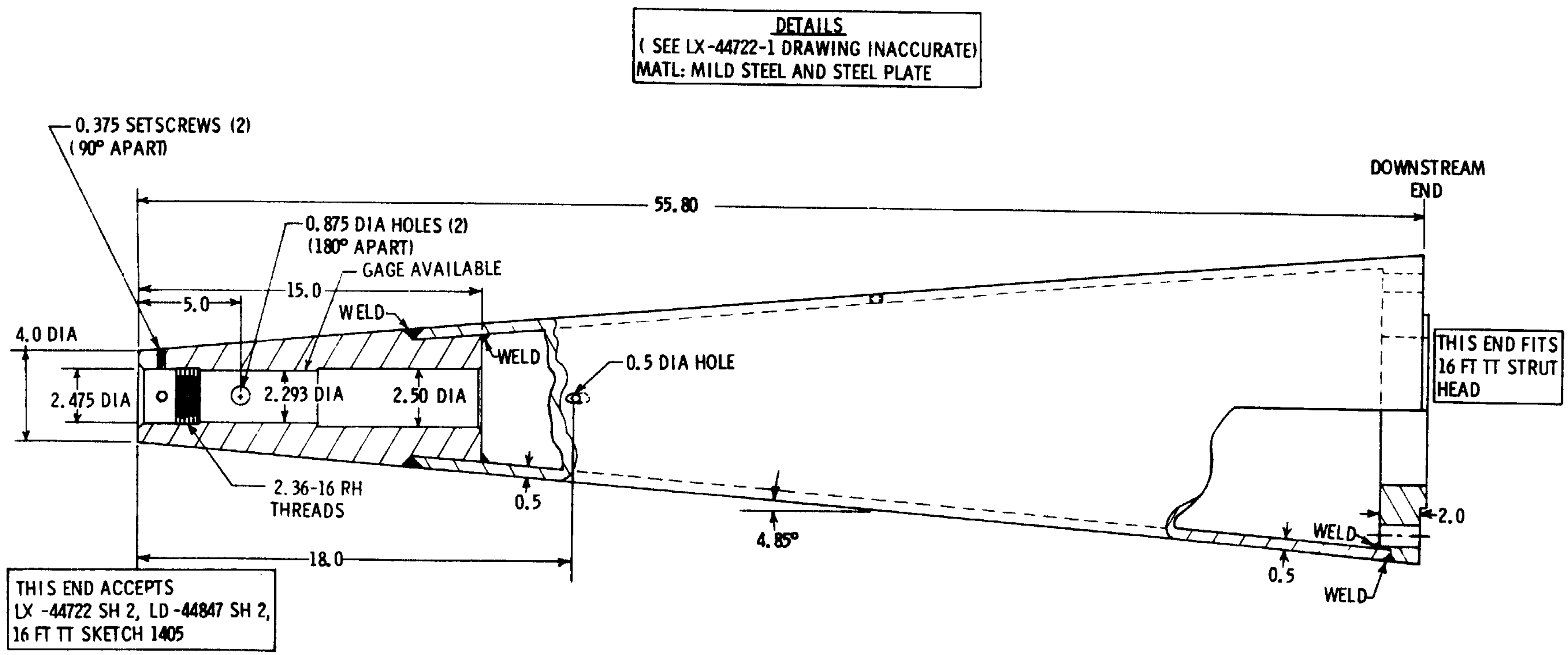


Figure II-9. Sketch of long sting cone LX-44722-1. (All dimensions are in inches unless otherwise indicated.)

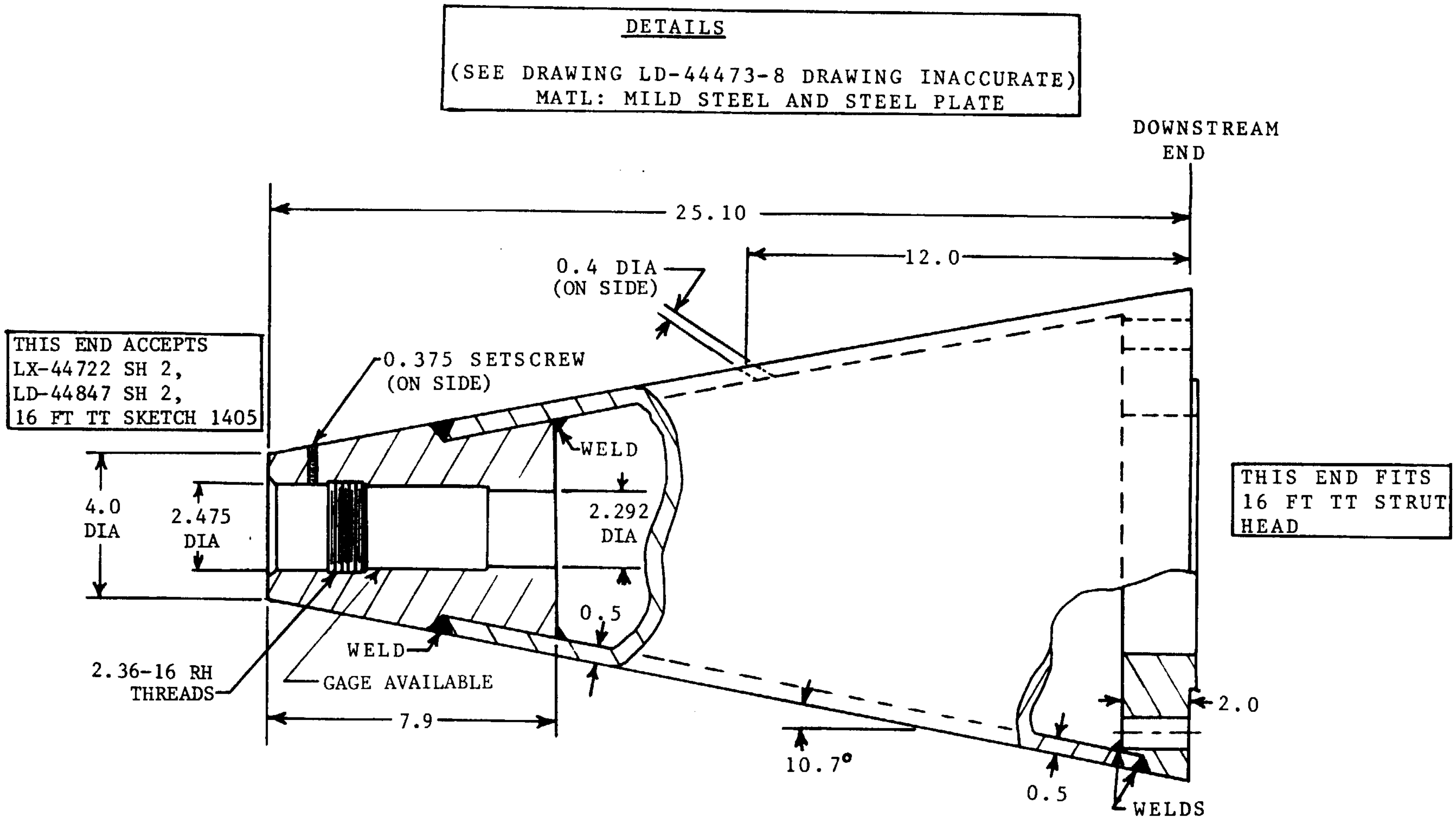


Figure II-10. Sketch of short sting cone LD-44473-8. (All dimensions are in inches unless otherwise indicated.)

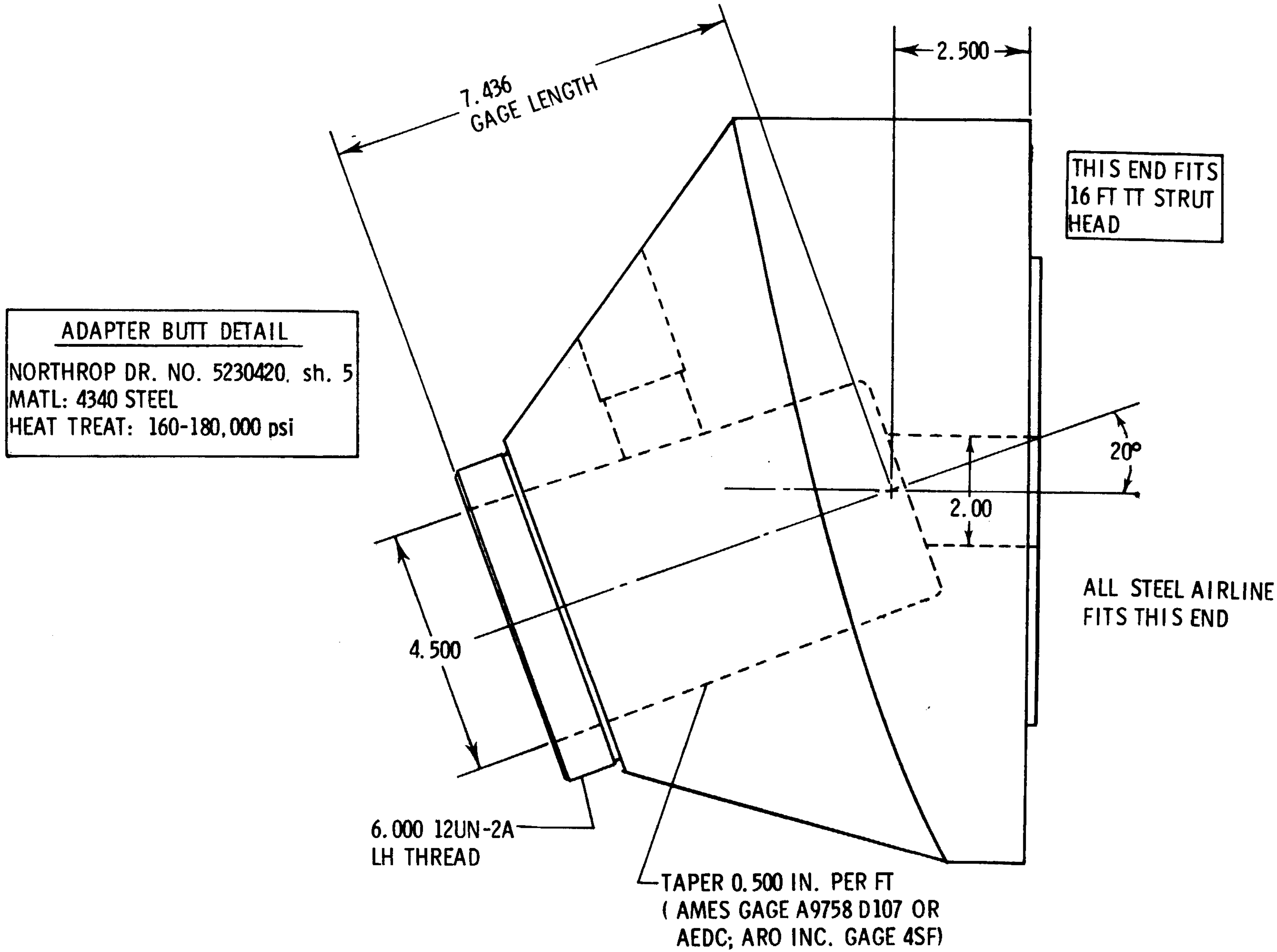
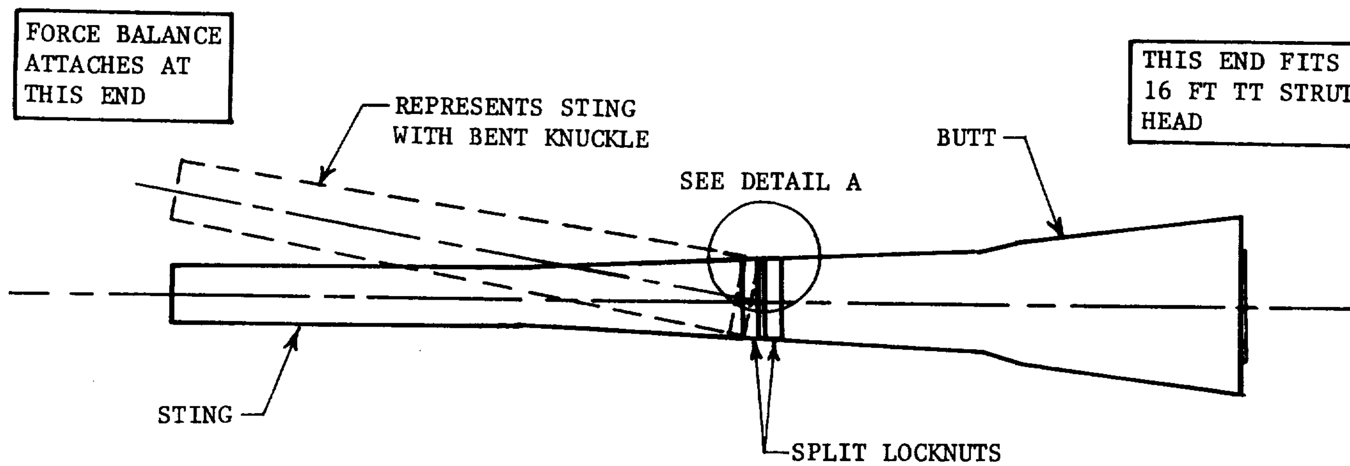
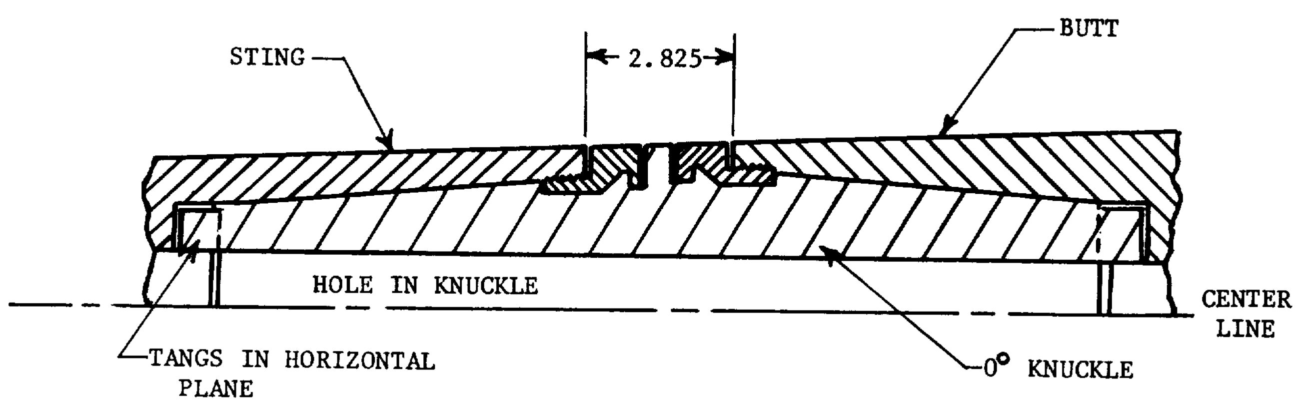


Figure II-11. Sketch of adapter butt 5230420 SH.5. (All dimensions are in inches unless otherwise indicated.)

(SEE DRAWING LD - 44571 SH.1)

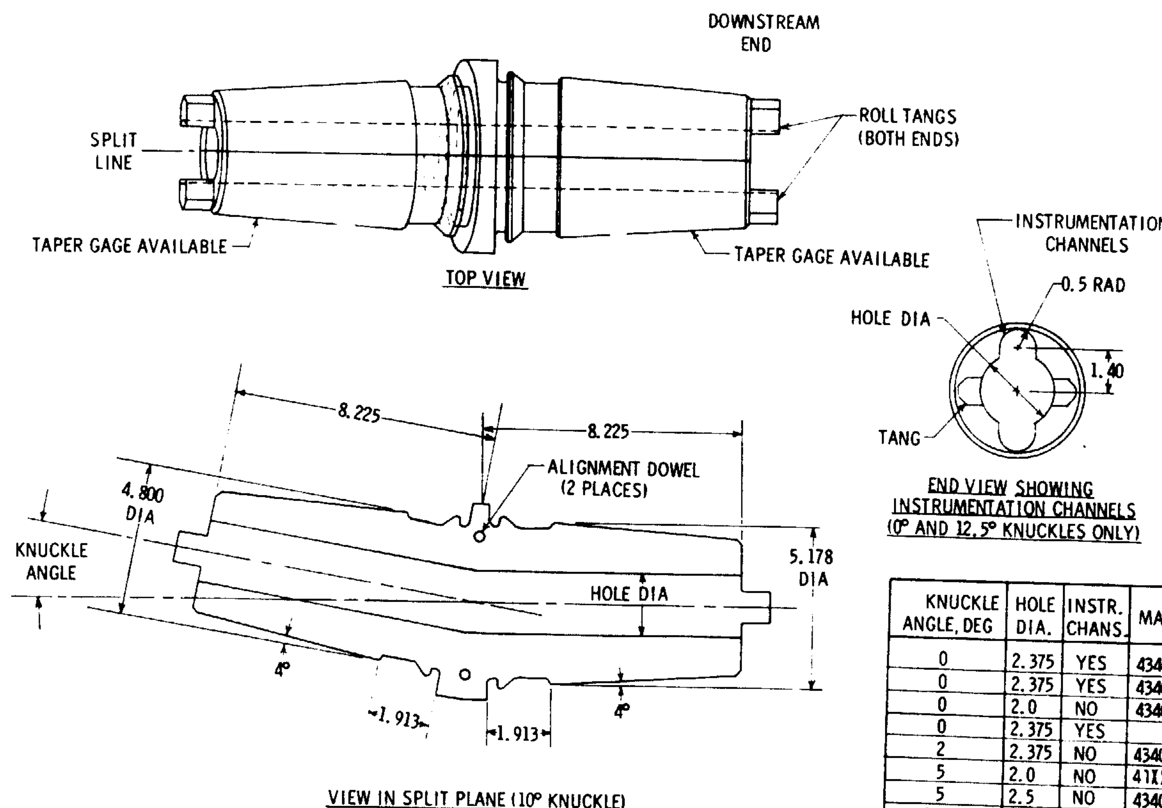




DETAIL A

CROSS SECTION OF TYPICAL ASSEMBLY. STING-0° KNUCKLE- BUTT. (NO ADAPTER)

Figure II-12. Sketch and cross section showing a typical sting-knuckle-butt assembly. (All dimensions are in inches unless otherwise indicated.)



KNUCKLE ANGLE, DEG	HOLE DIA.	INSTR. CHANS.	MATERIAL	MIN. YIELD, PSI	DRAWING NUMBER
0	2.375	YES	4340 STL	180,000	GRUMMAN 303MOD4135
0	2.375	YES	4340 STL	180,000	GRUMMAN 303MOD4135
0	2.0	NO	4340 STL	150,000	LD - 44571 SH. 6
0	2.375	YES		150,000	(LABELED MILD STEEL)
2	2.375	NO	4340 STL	150,000	LD - 44571
5	2.0	NO	41XX STL	130,000	LD - 44571
5	2.5	NO	4340 STL	180,000	ROHR 501-093
10	2.0	NO	4340 STL	150,000	LD - 44571 SH. 4
12.5	2.375	YES	4340 STL	180,000	LD - 534207
15	2.0	NO		150,000	LD - 44571
20	2.0	NO			LD - 44571
30	2.375	NO			LD - 44571
45	2.375	NO			LD - 44571
10 UP 5 RIGHT	2.5	NO	4340 STL	180,000	ROHR 501-109 FILED AS LRC DWG AT BLDG1146

Figure II-13. Sketch of typical knuckles available. (All dimensions are in inches unless otherwise indicated.)

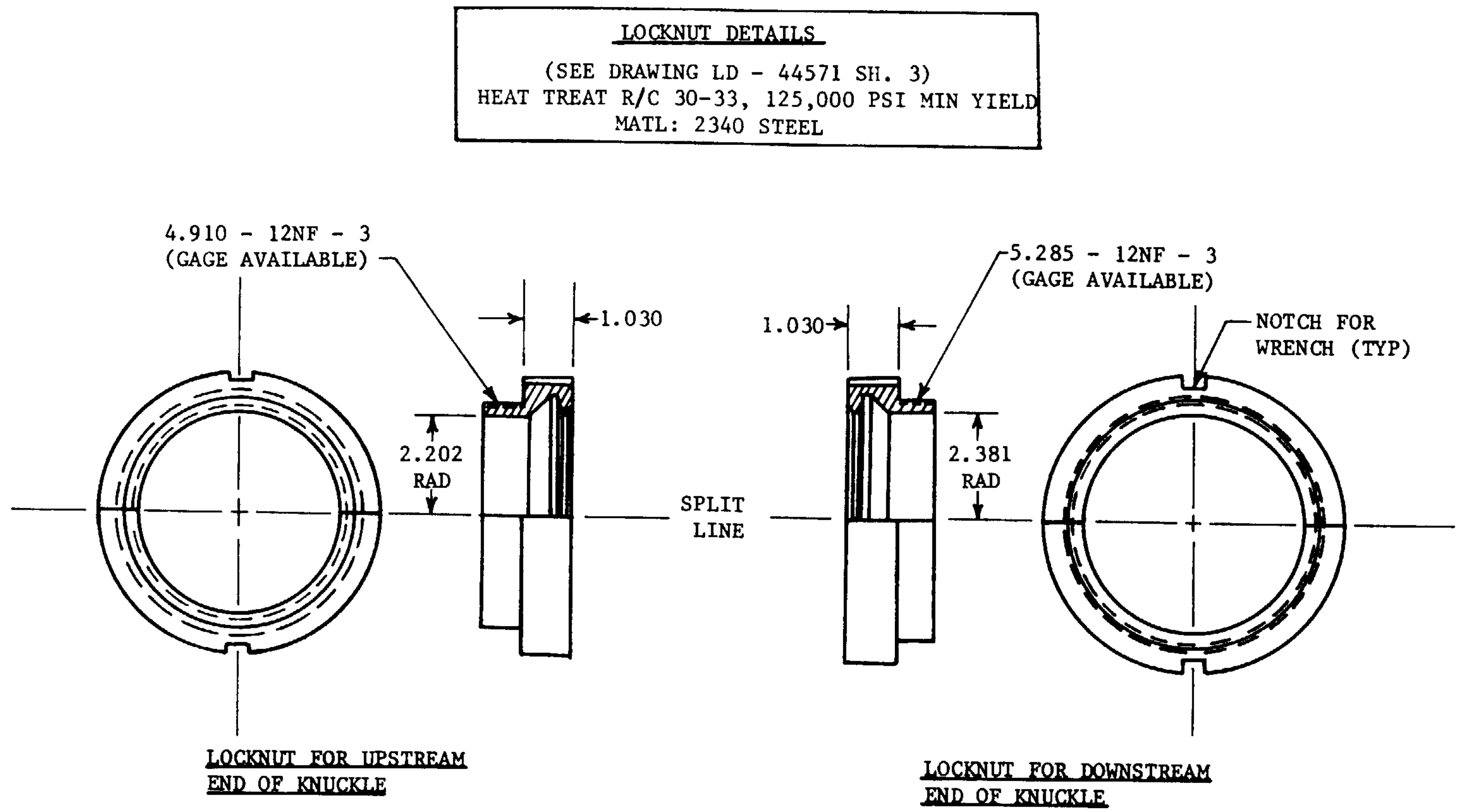


Figure II-14. Sketch of split locknuts used to seat knuckle tapers. (All dimensions are in inches unless otherwise indicated.)

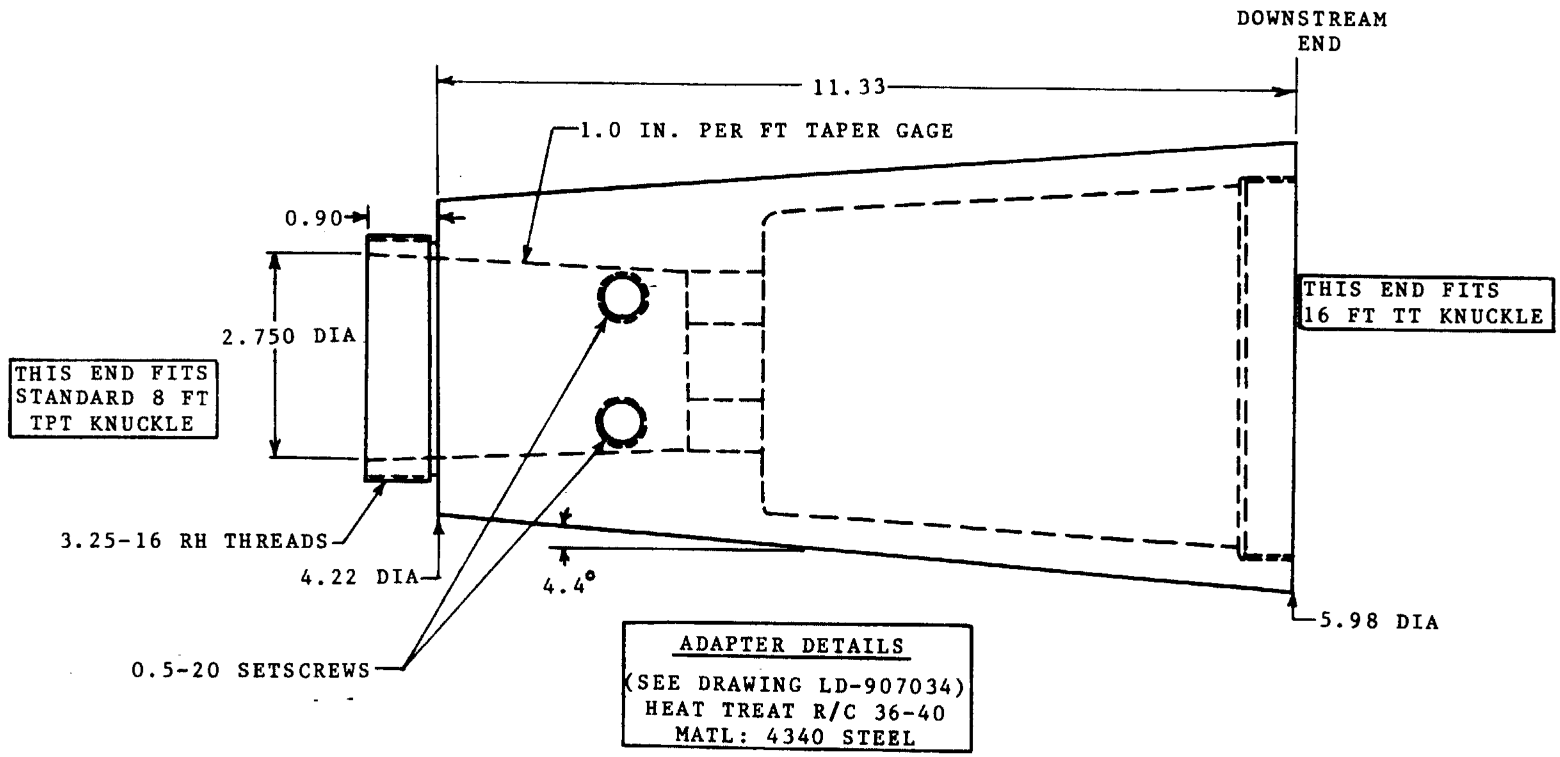


Figure II-15. Sketch of adapter LD-907034. (All dimensions are in inches unless otherwise indicated.)

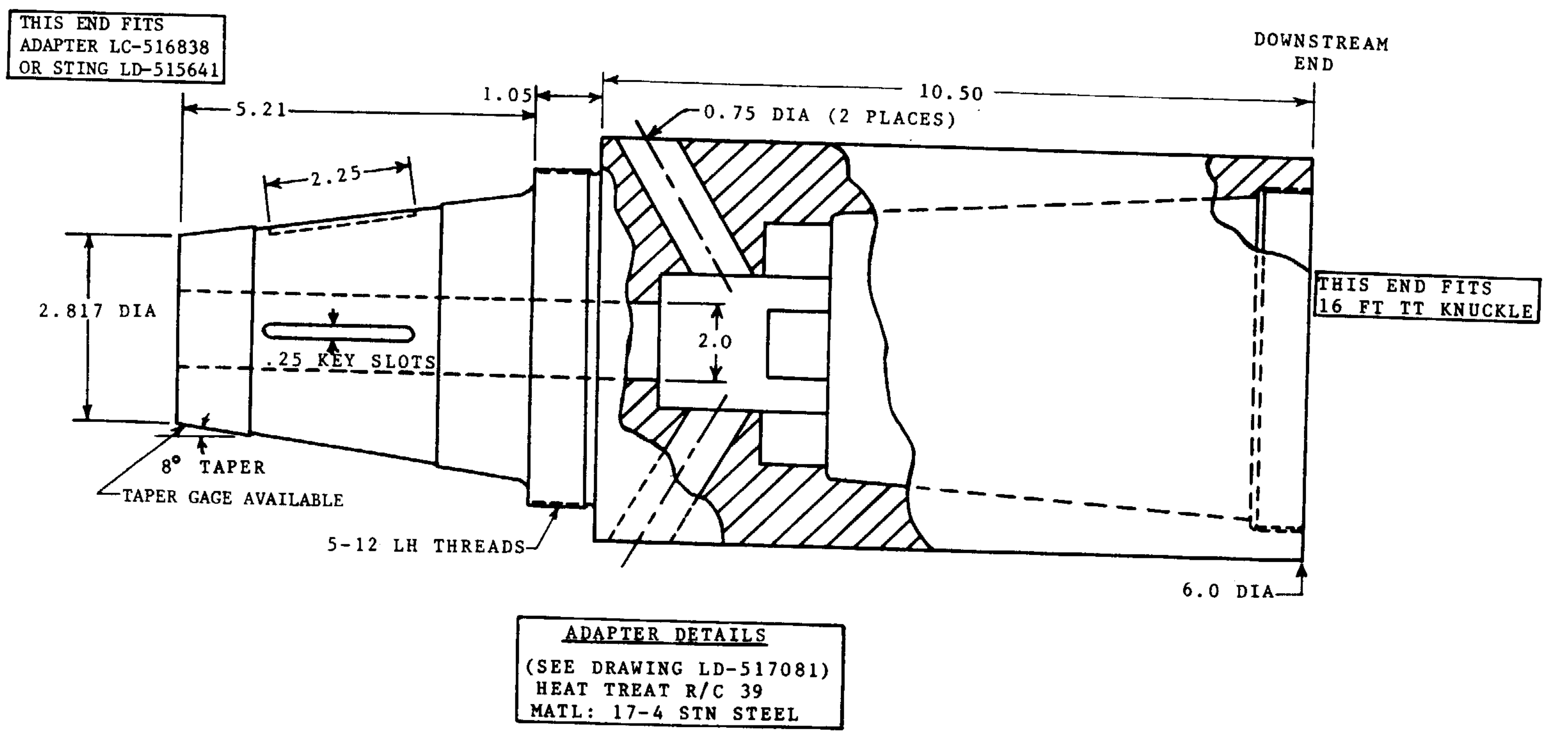


Figure II-16. Sketch of Adapter LD-517081A. (All dimensions are in inches unless otherwise indicated.)

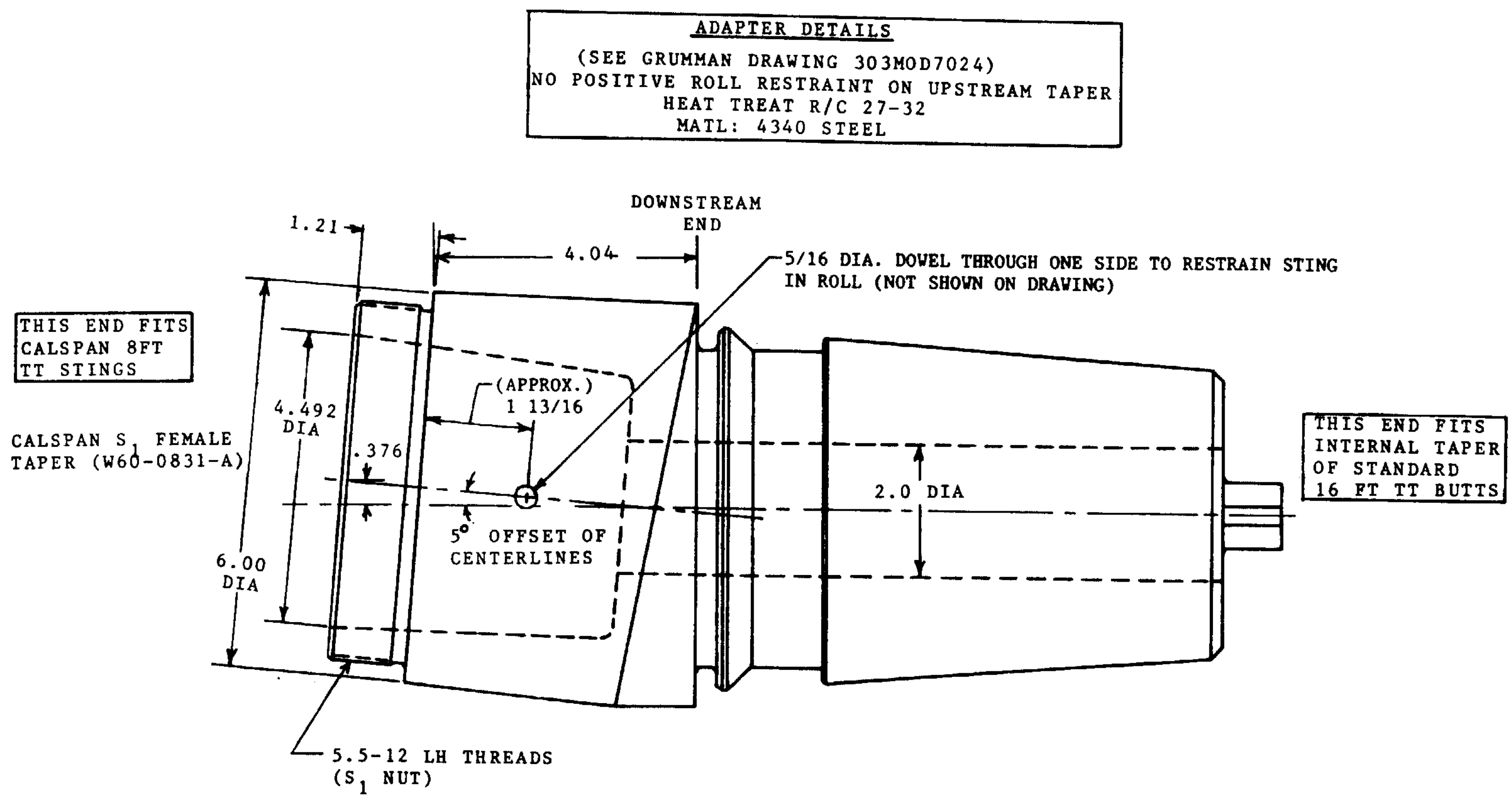


Figure II-17. Sketch of adapter 303MOD7024. (All dimensions are in inches unless otherwise indicated.)

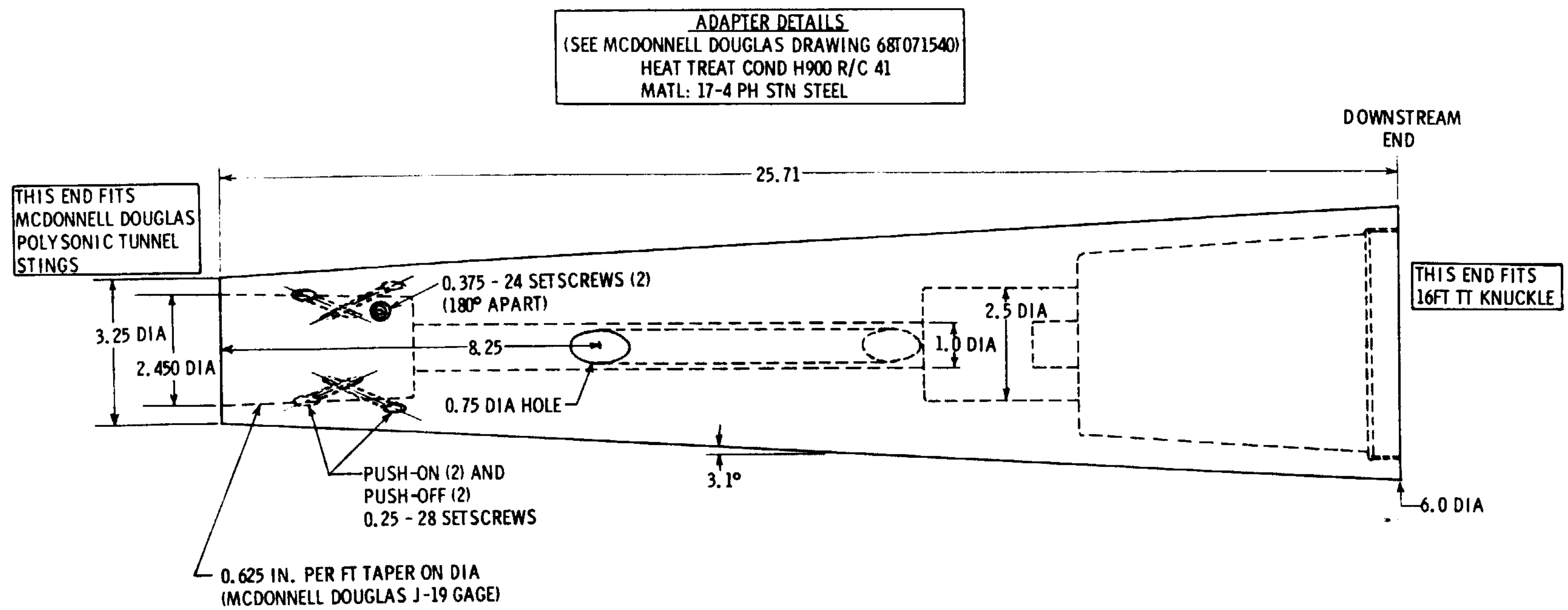


Figure II-18. Sketch of adapter 68T071540. (All dimensions are in inches unless otherwise indicated.)

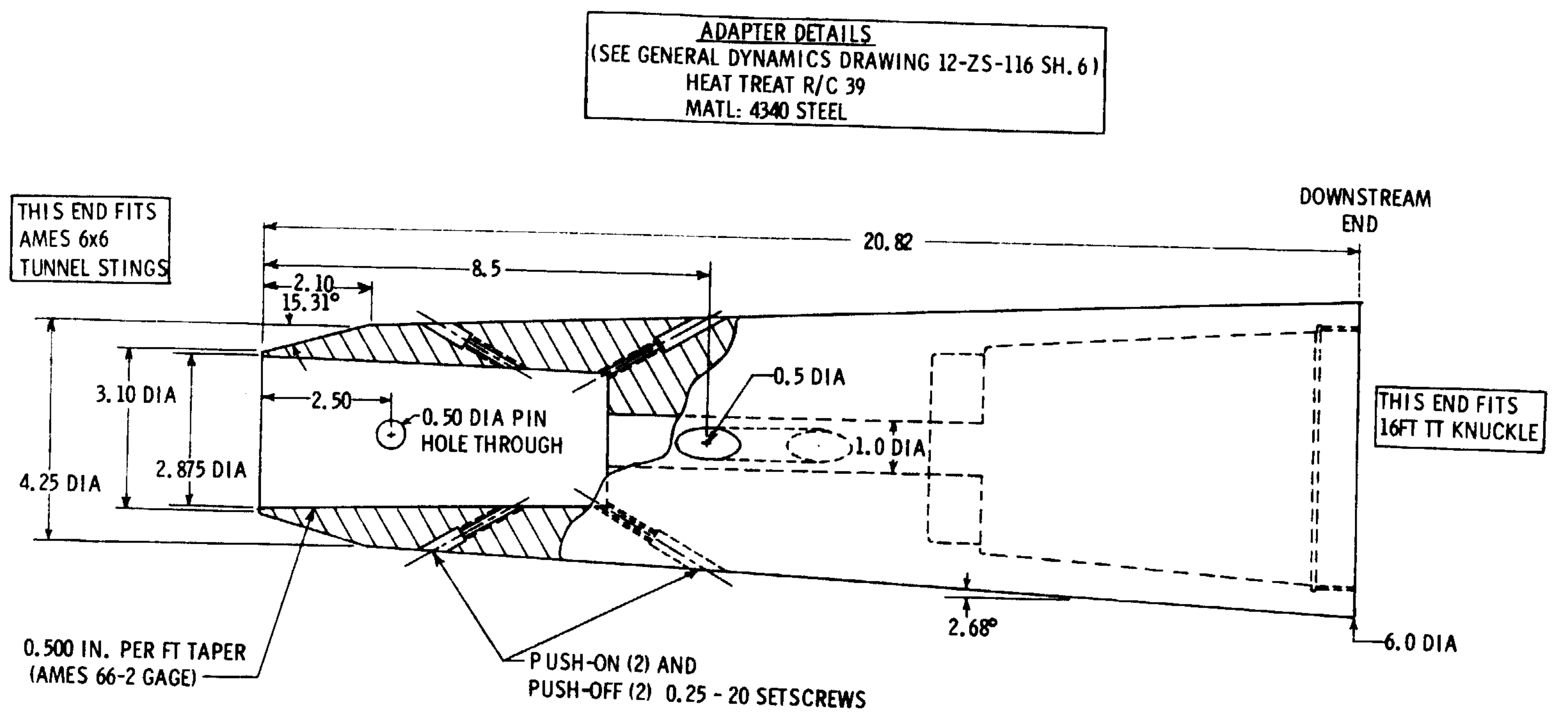


Figure II-19. Sketch of adapter 12-ZS-116 SH.6. (All dimensions are in inches unless otherwise indicated.)

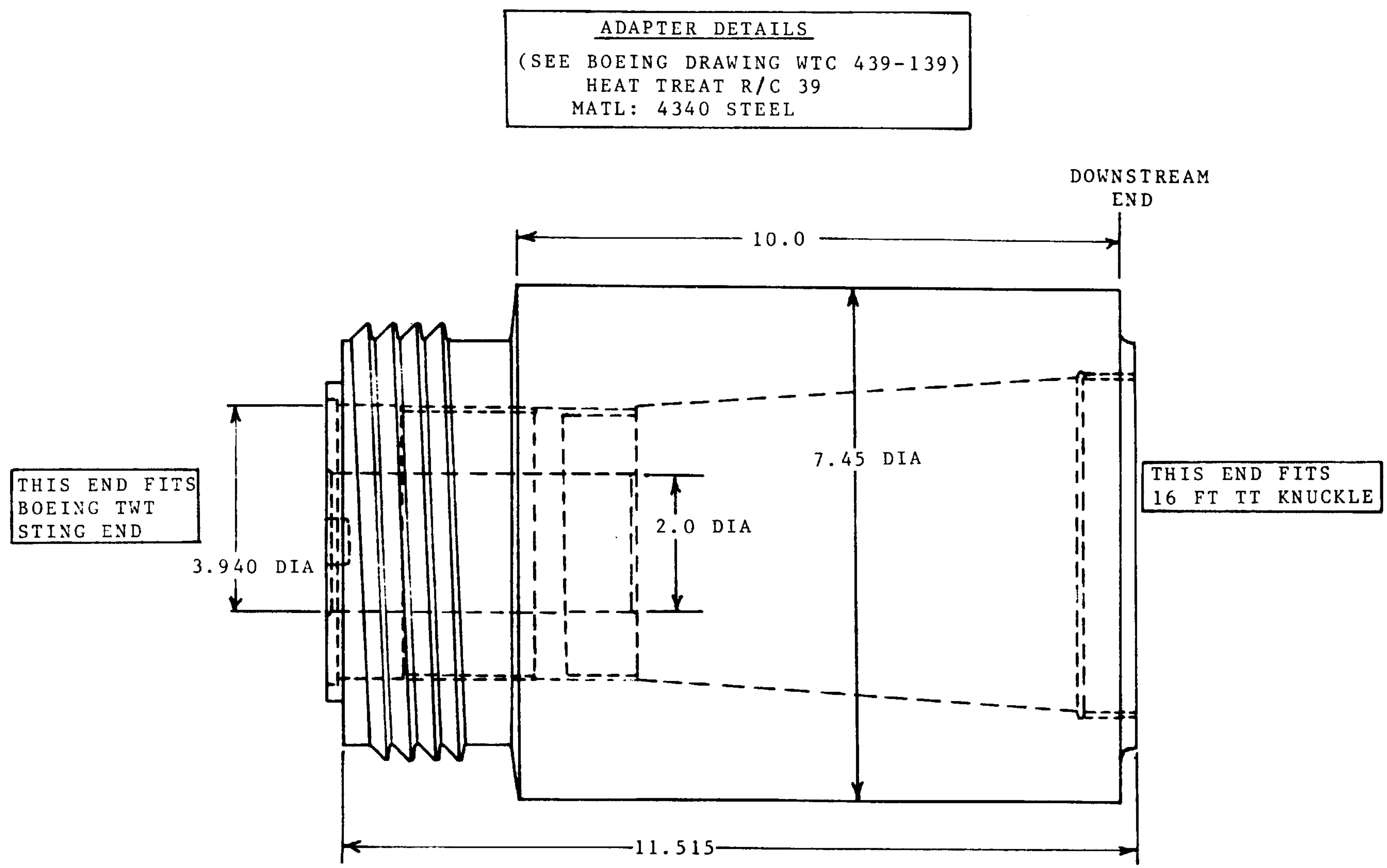


Figure II-20. Sketch of adapter WTC 439 - 139. (All dimensions are in inches unless otherwise indicated.)

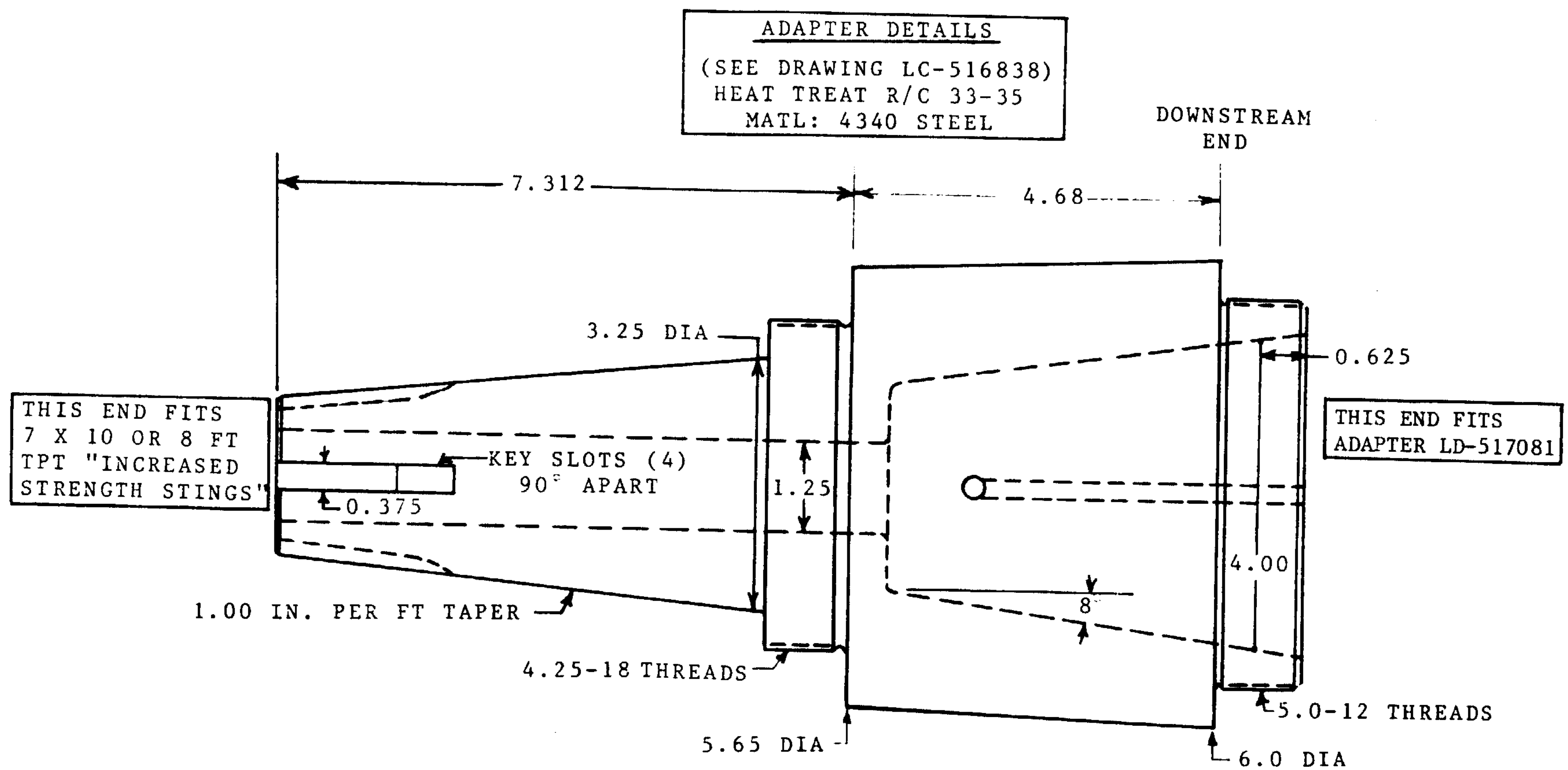


Figure II-21. Sketch of adapter LC - 516838. (All dimensions are in inches unless otherwise indicated.)

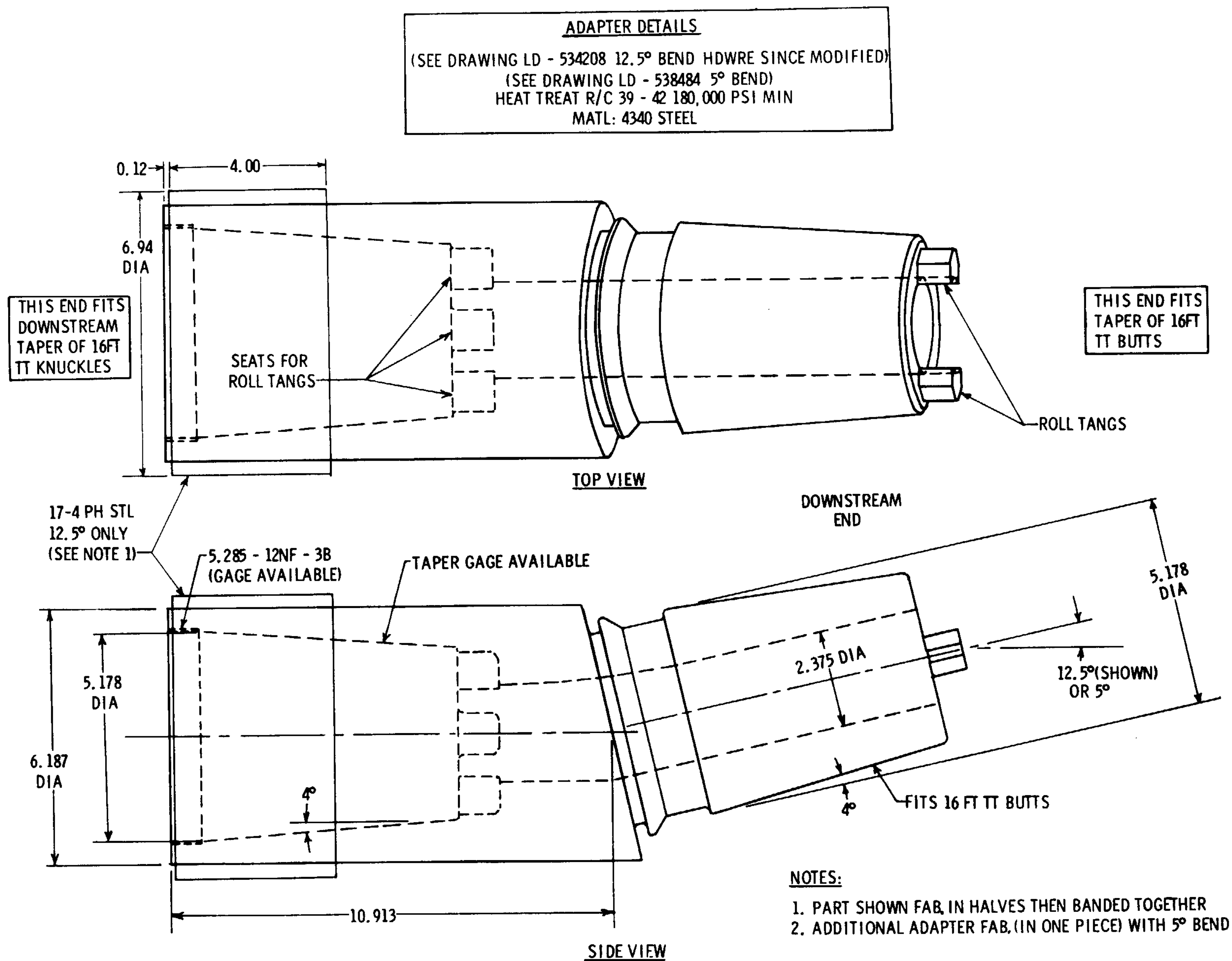


Figure II-22. Sketch of knuckle adapters LD-534208 and LD-538484. (All dimensions are in inches unless otherwise indicated.)

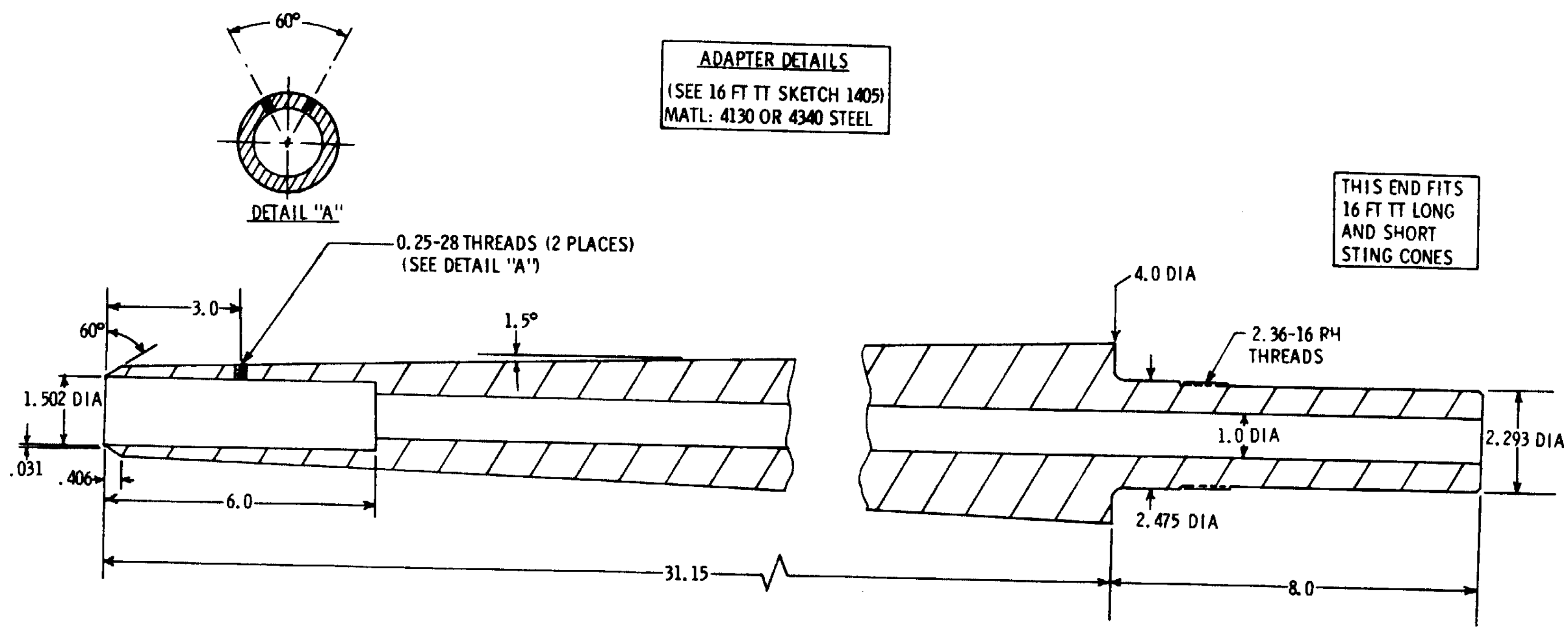


Figure II-23. Sketch of adapter SKETCH 1405. (All dimensions are in inches unless otherwise indicated.

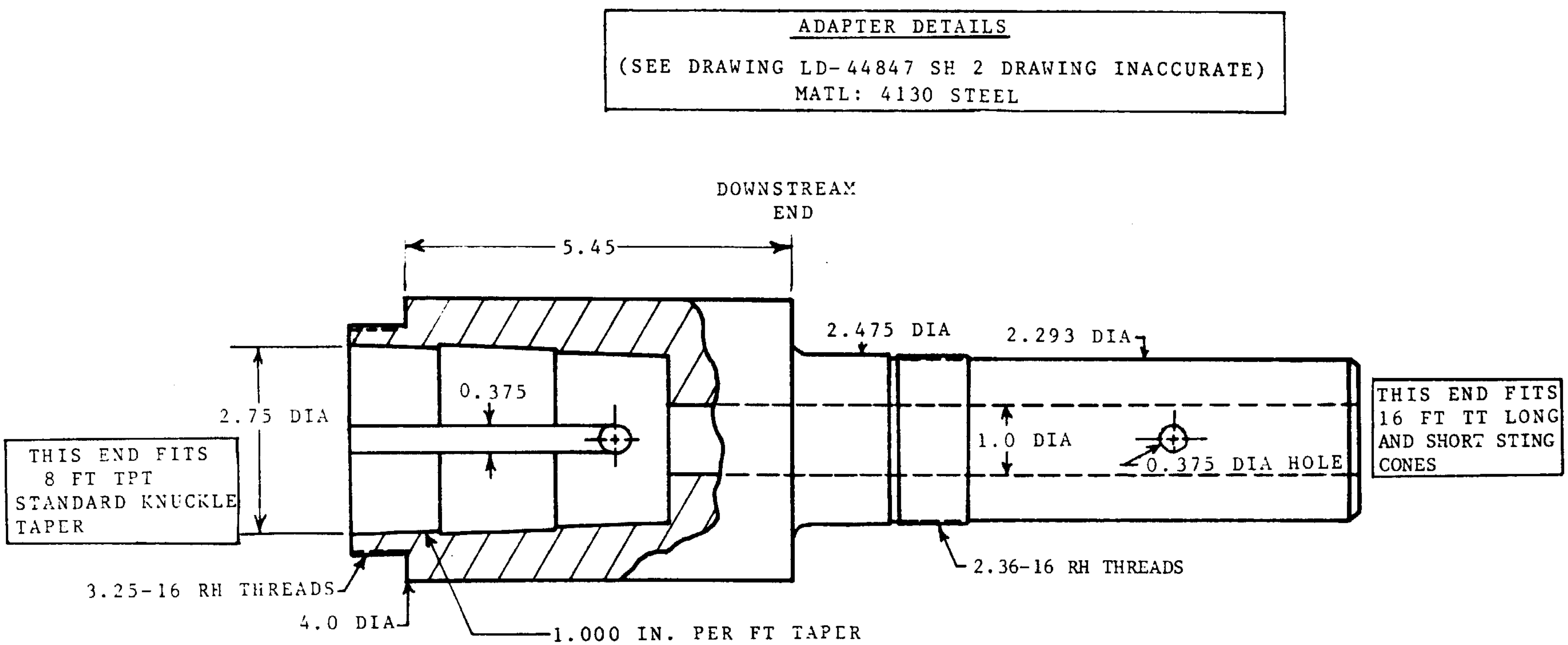


Figure II-24. Sketch of adapter LD-44847 SH. 2. (All dimensions are in inches unless otherwise indicated.)

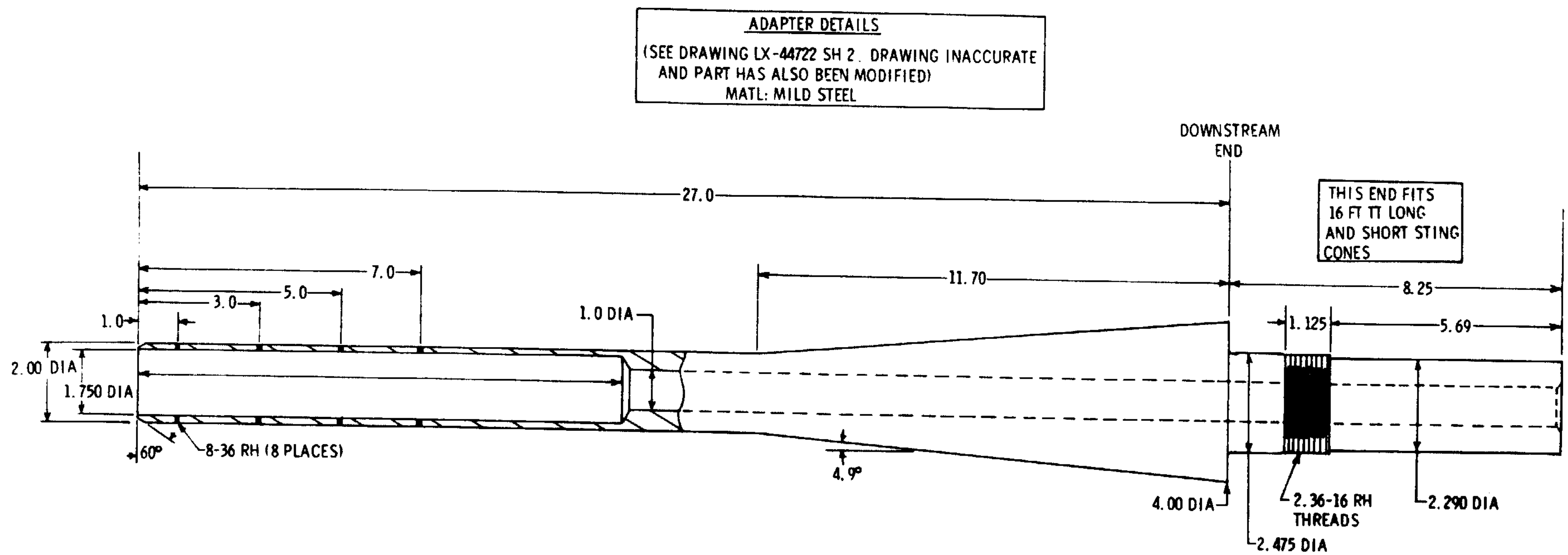


Figure II-25. Sketch of adapter LX-44722 SH.2. (All dimensions are in inches unless otherwise indicated.)

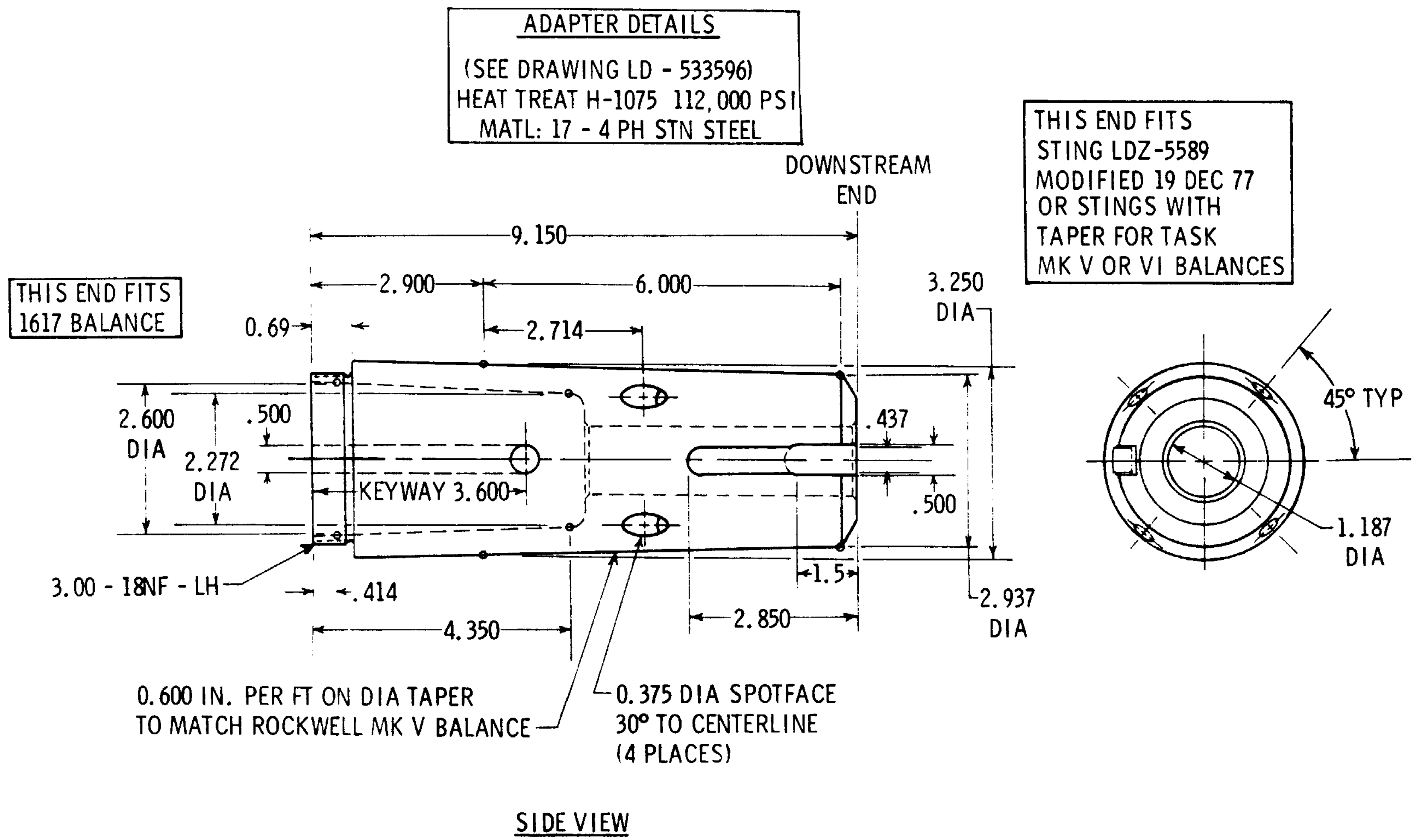


Figure II-26. Sketch of balance-to-sting adapter LD-533596. (All dimensions are in inches unless otherwise indicted.)

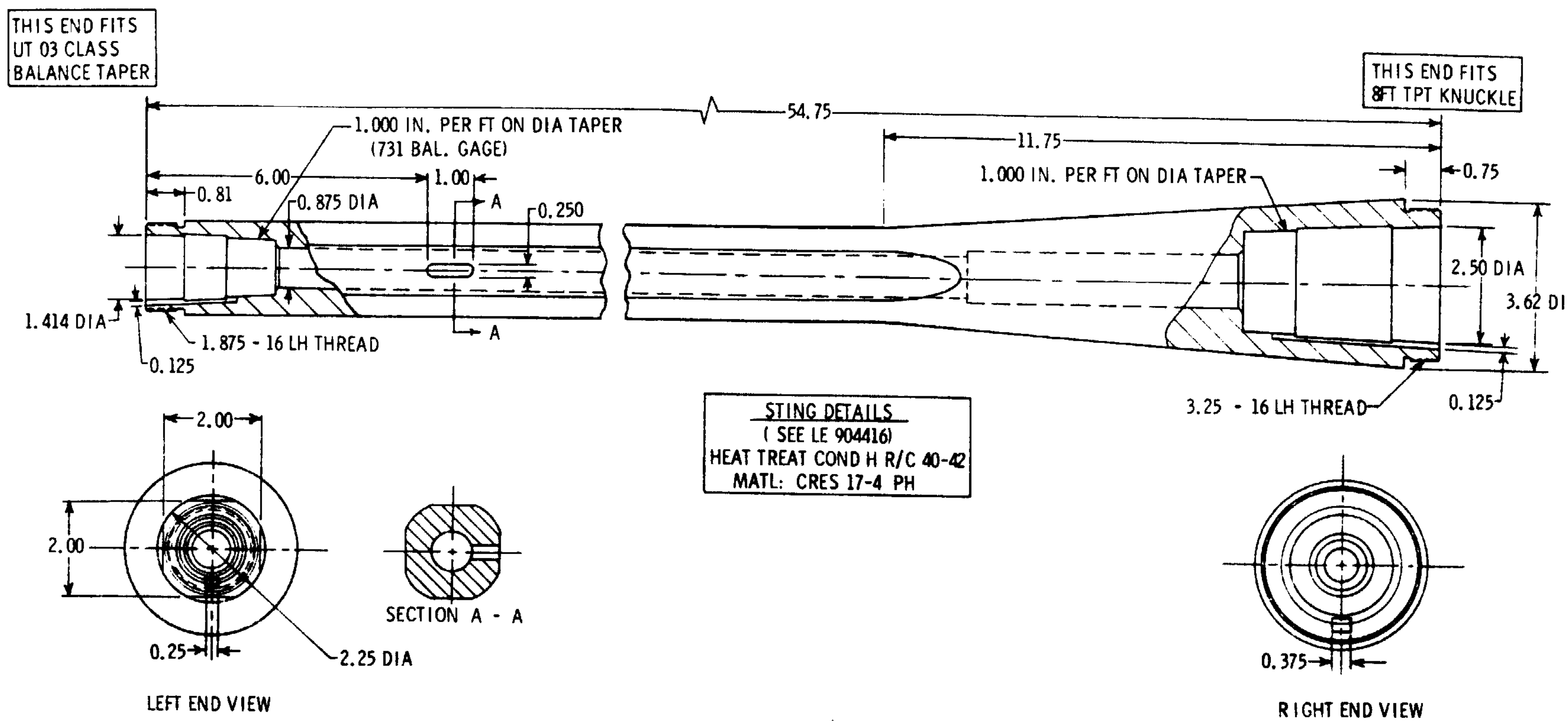


Figure II-27. Sketch of sting LE-904416. (All dimensions are in inches unless otherwise indicated.)

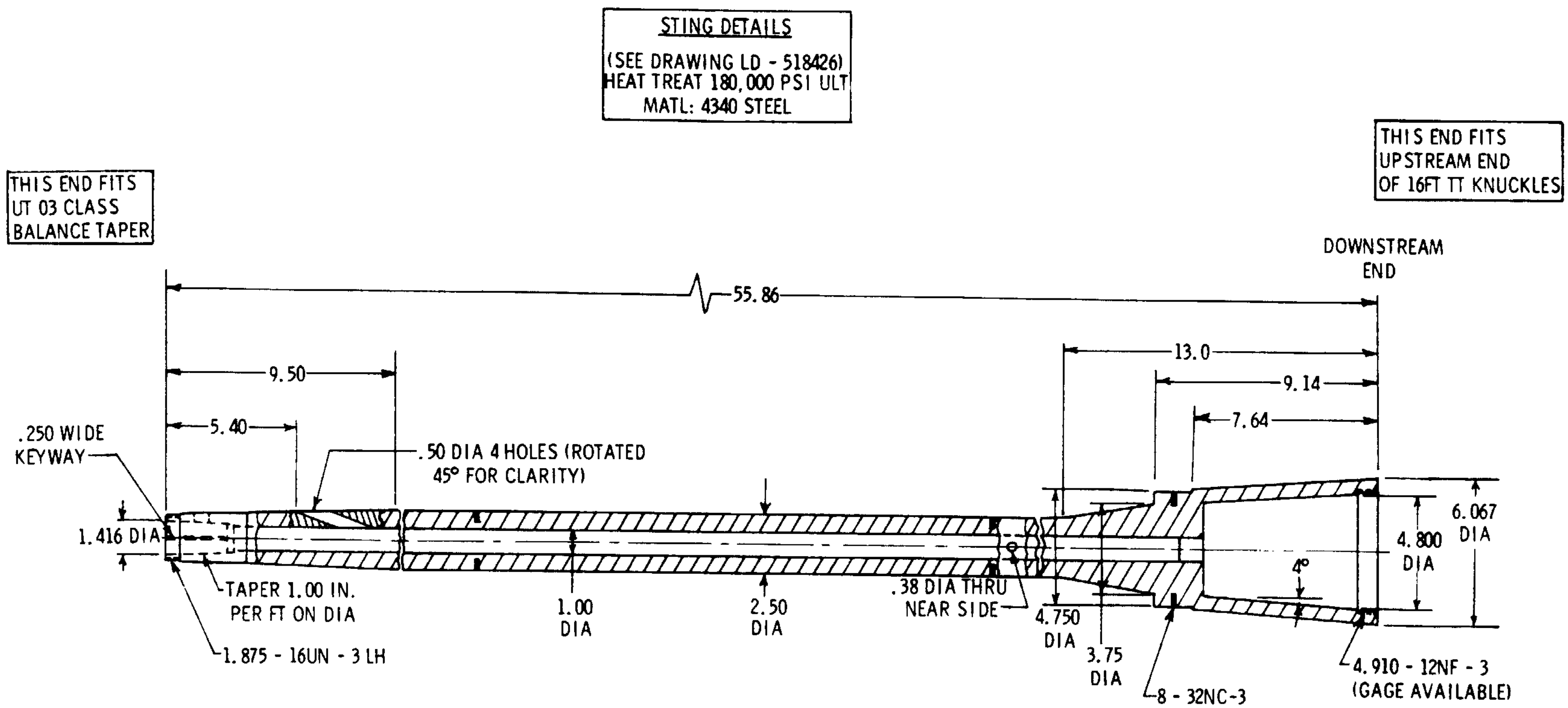


Figure II-28. Sketch of sting LD-518426. (All dimensions are in inches unless otherwise indicated.)

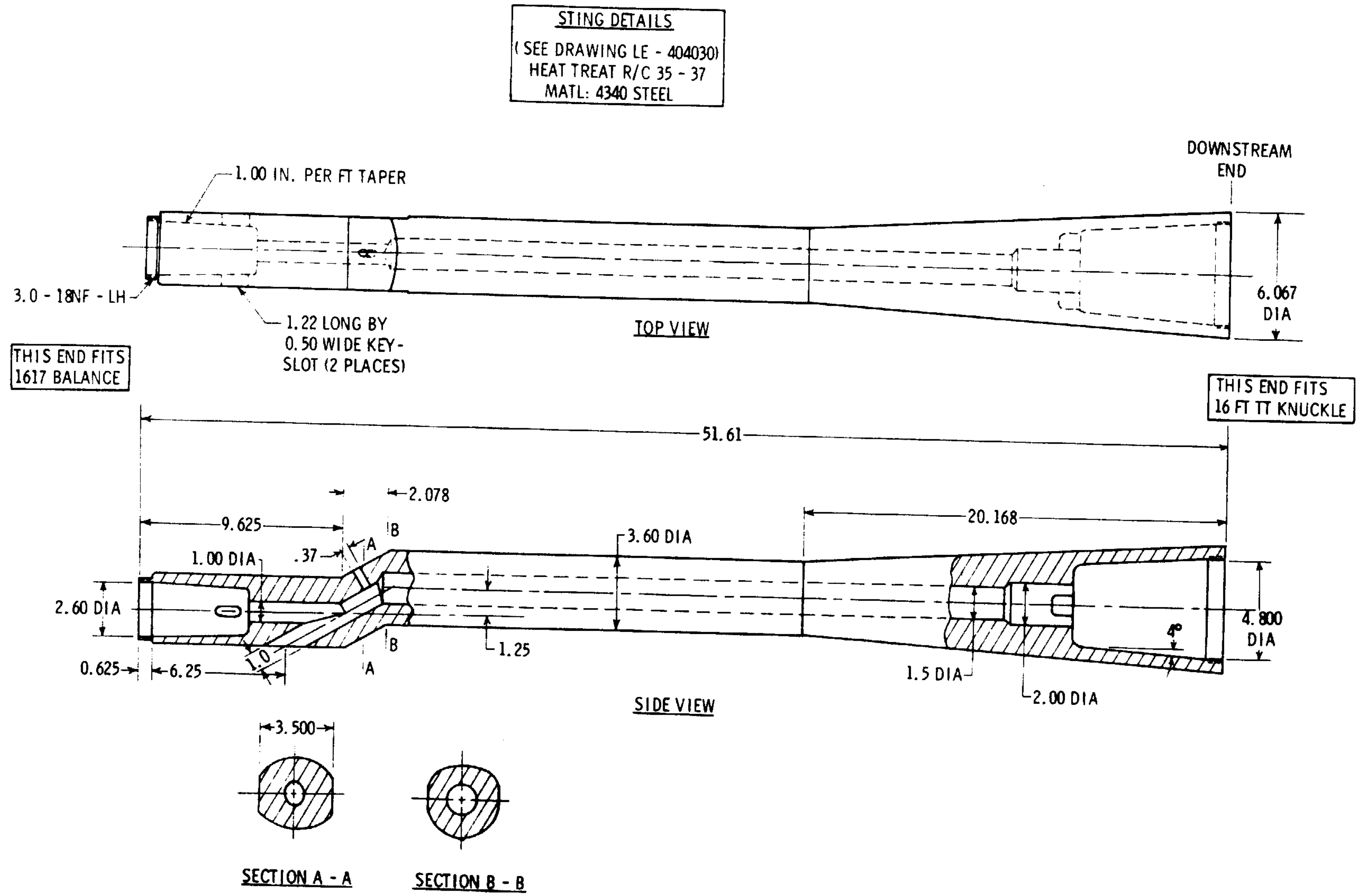


Figure II-29. Sketch of sting LE-404030. (All dimensions are in inches unless otherwise indicated.)

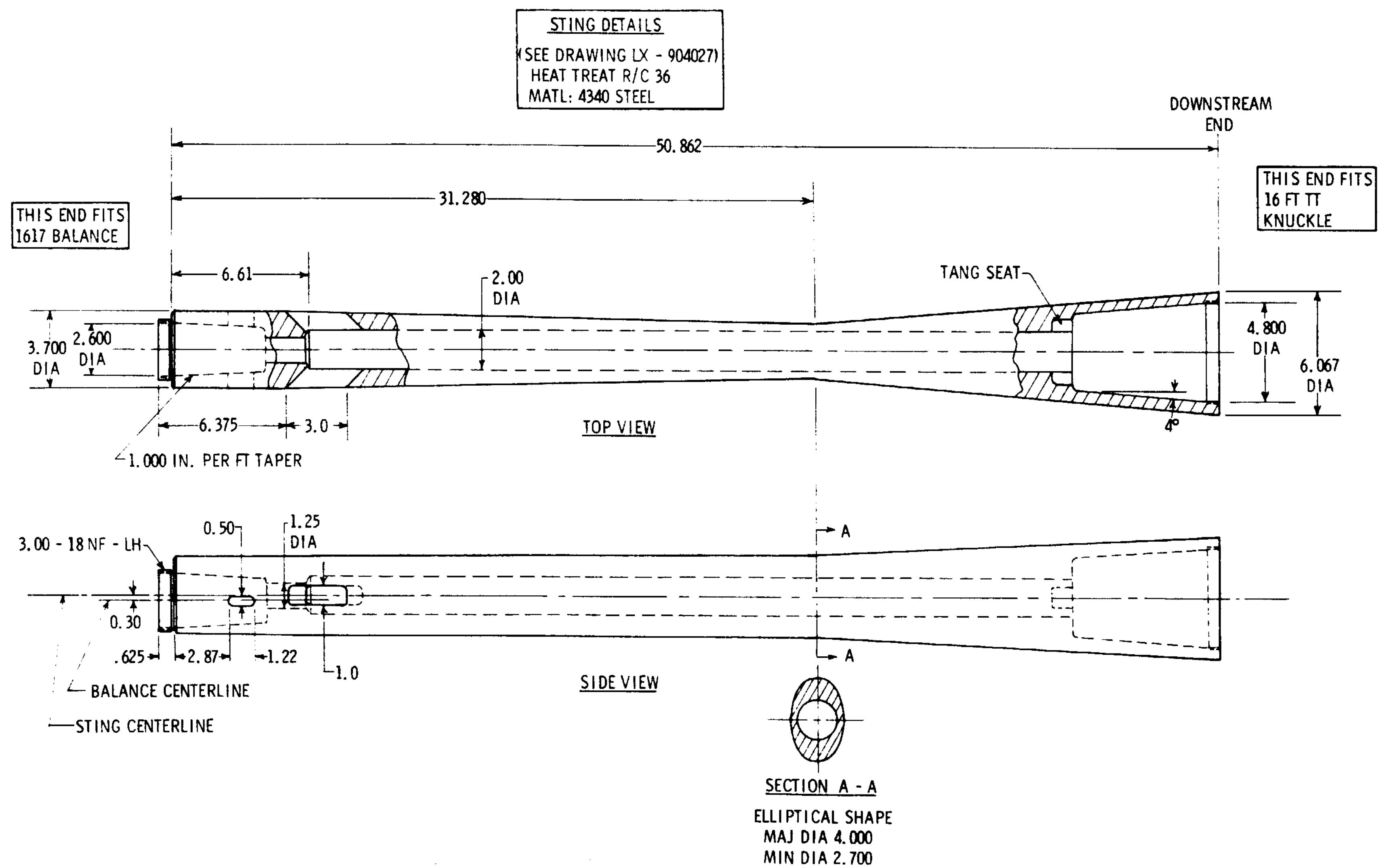


Figure II-30. Sketch of sting LX-904027. (All dimensions are in inches unless otherwise indicated.)

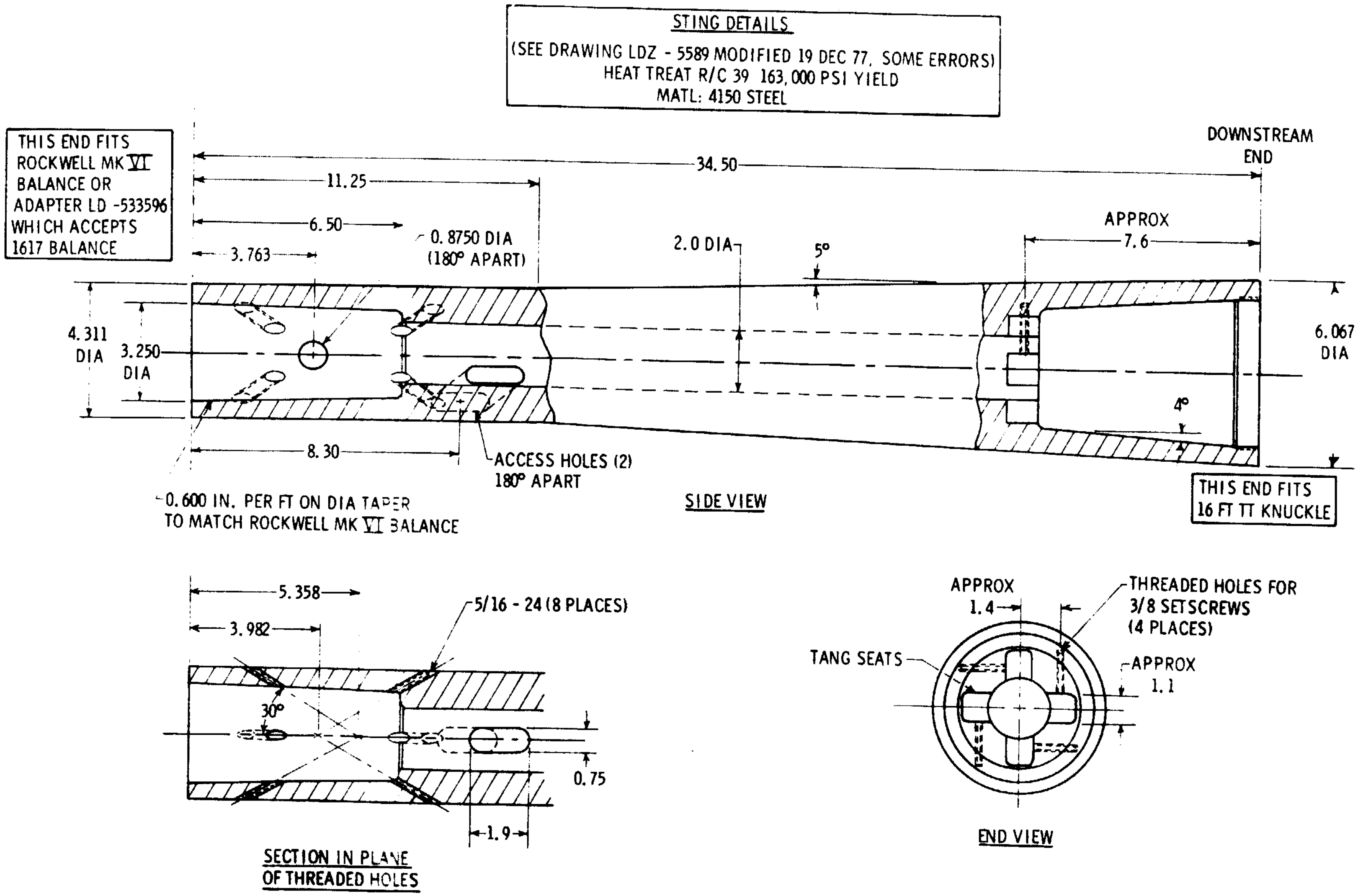


Figure II-31. Sketch of sting LDZ-5589 MODIFIED 19 DEC 77. (All dimensions are in inches unless otherwise indicated.)

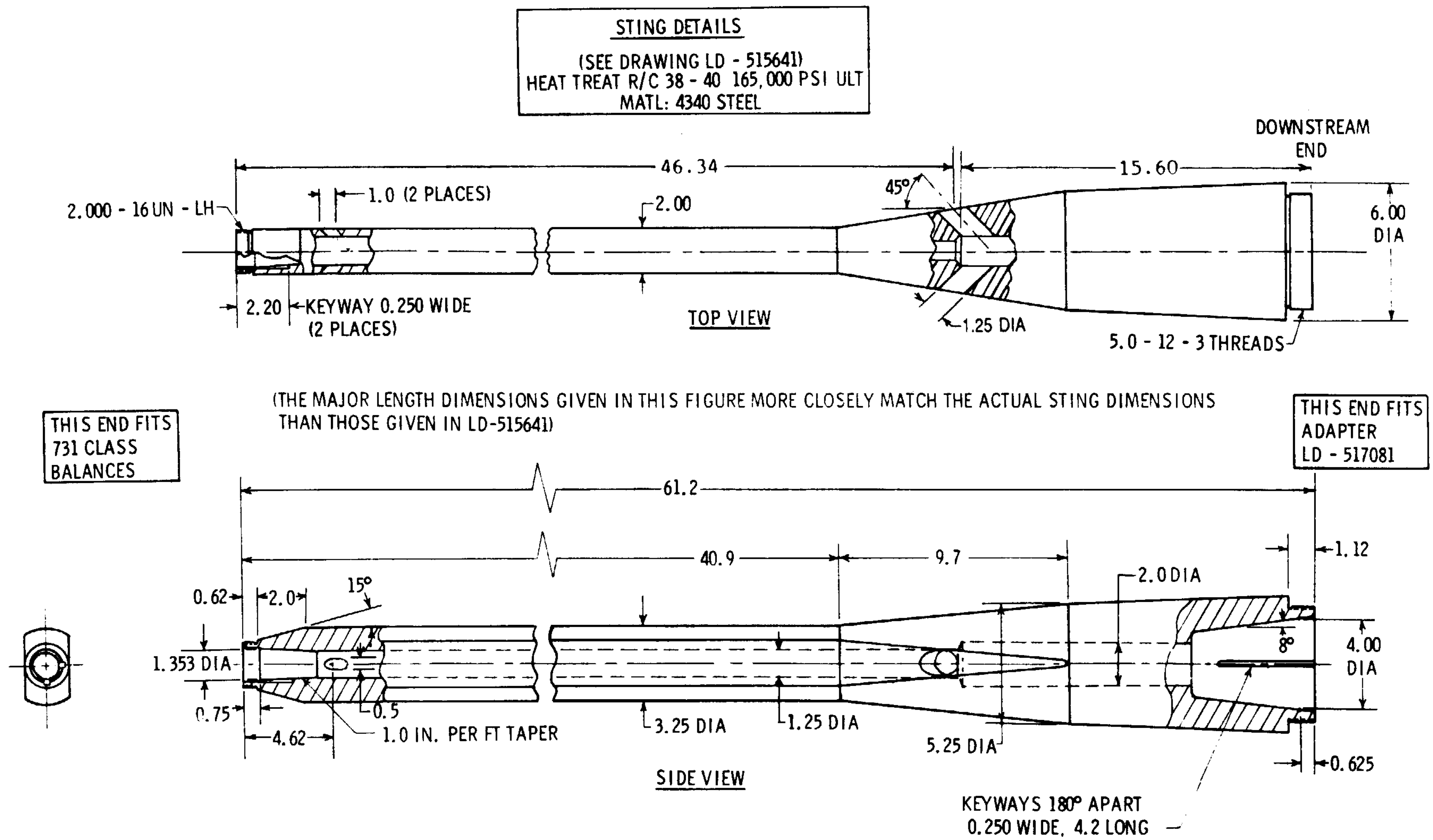


Figure II-32. Sketch of sting LD-515641A. (All dimensions are in inches unless otherwise indicated.)

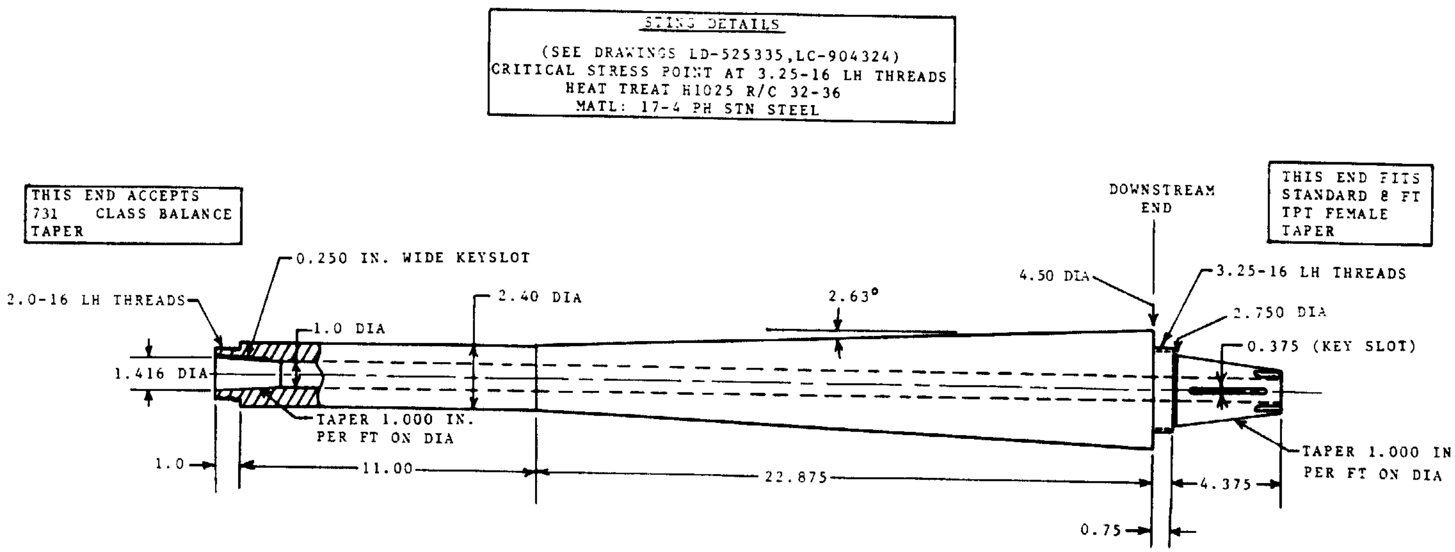


Figure II-33. Sketch of sting LD-525335. (All dimensions are in inches unless otherwise indicated.)

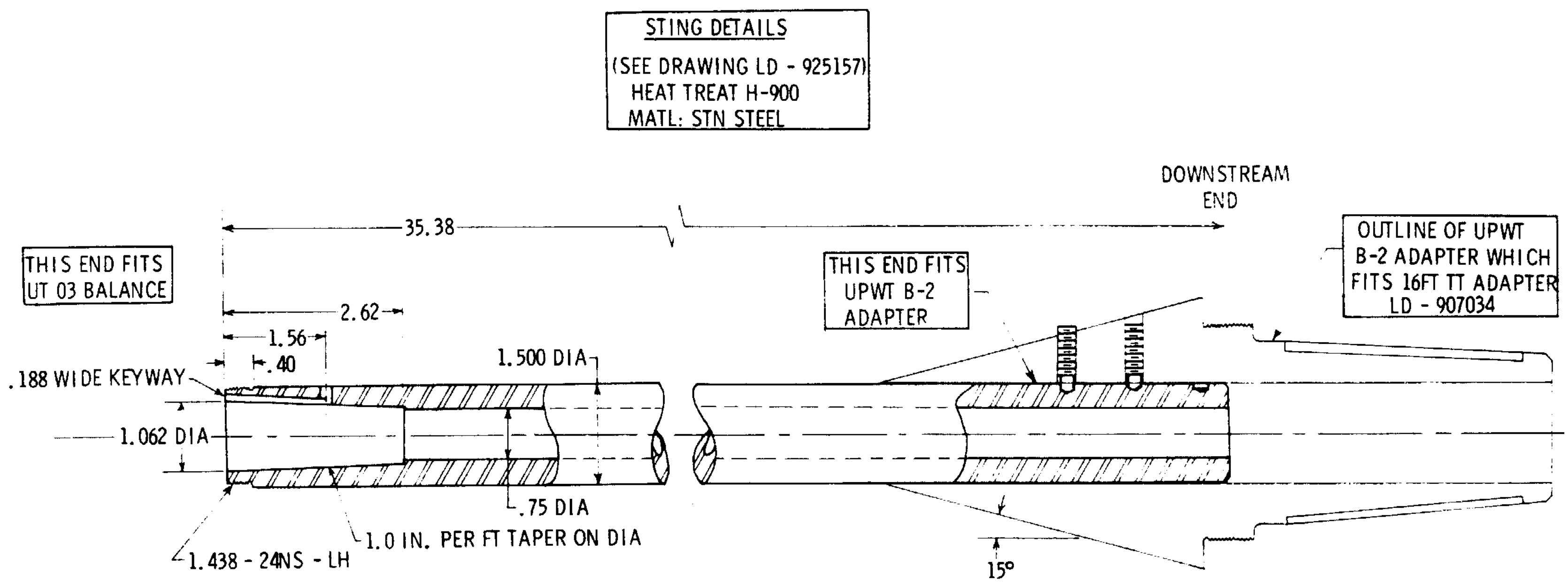


Figure II-34. Sketch of sting LD-925157. (All dimensions are in inches unless otherwise indicated.)

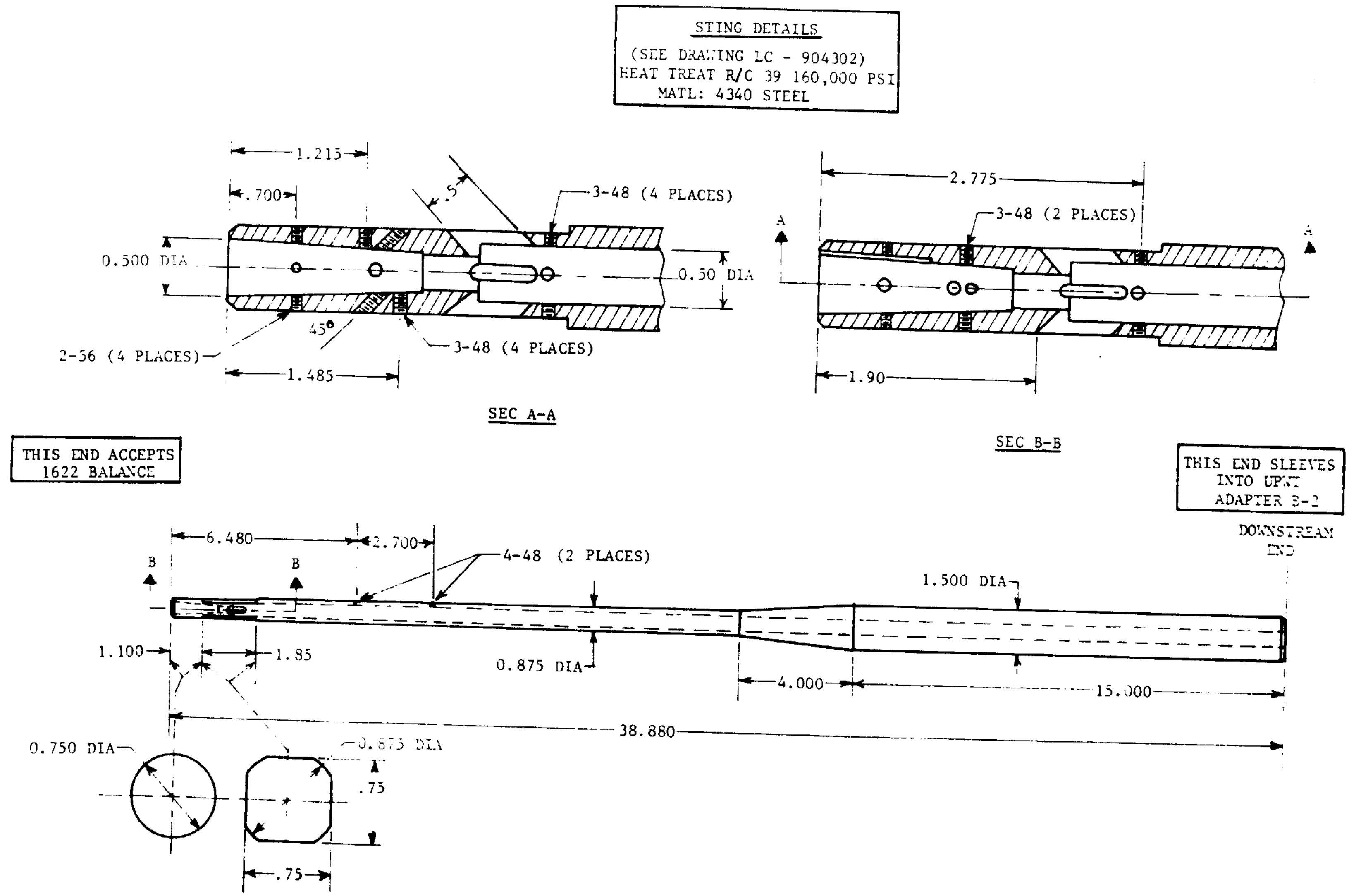


Figure II-35. Sketch of sting LC-904302. (All dimensions are in inches unless otherwise indicated.)

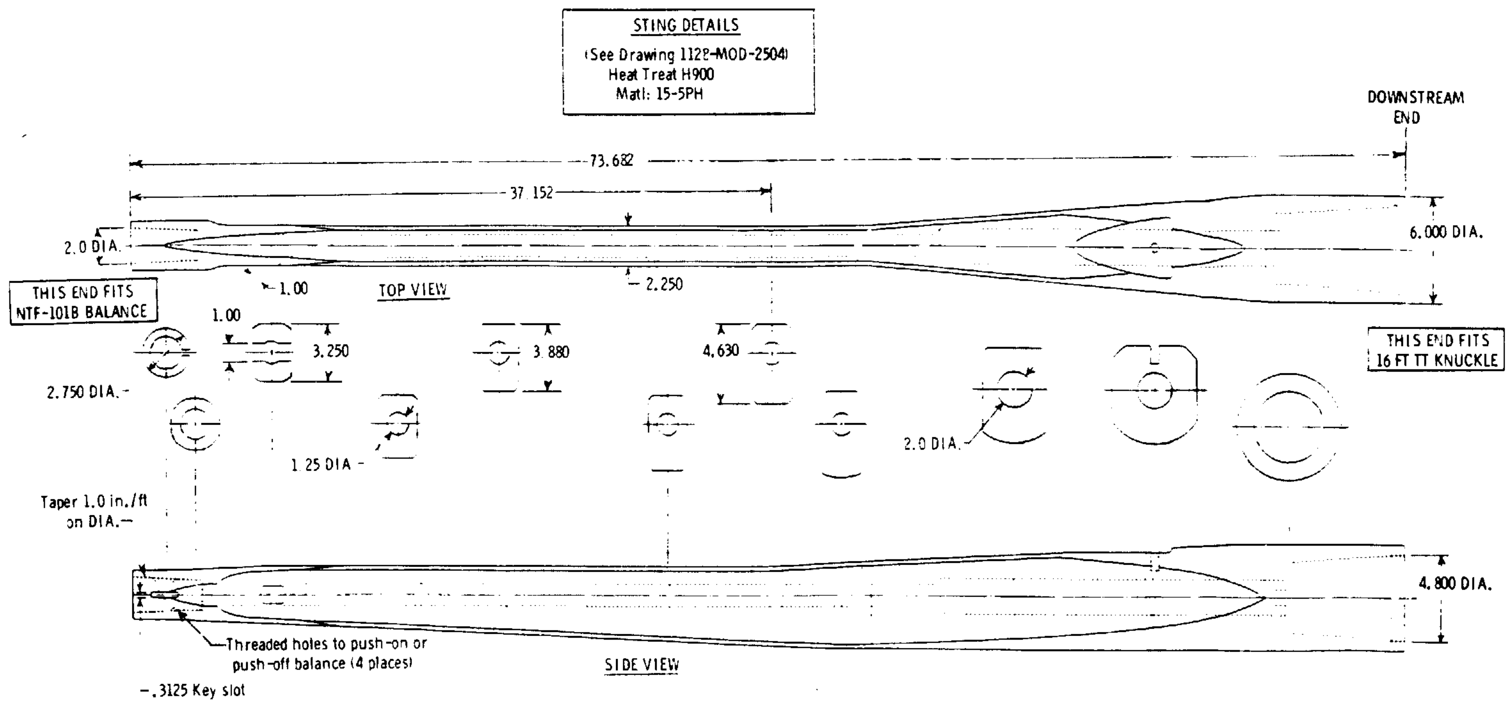


Figure II-36. Sketch of sting 1128-MOD-2504 (Grumman). (All dimensions are in inches.)

SECTION III - Model Support System for Propulsion Simulation Testing

The 16-Foot Transonic Tunnel support system is shown in figure III-1 (identical to figure II-1). All model support hardware attach to this system at either the forward end of the strut head at Tunnel Station 141.94 or at the forward sting "butt" joint at Tunnel Station 138.33. As shown in figure III-2, two separate air supplies are provided internally within the strut head. Flexible high-pressure (1800 psi) air lines are used as a connection between the fixed air line in the strut head and the model. Each independently controlled air supply has the capability of providing an airflow of up to 15 lbs/sec at temperatures from 70° F to 130° F. The load capacity of the main support system is 10,000 lbs. of normal force and 10,000 lbs. of side force acting at Tunnel Station 134. The support system has an angle-of-attack range from -10° to 25°. A remote rotary control is located within the strut head so that the entire model, model support hardware, and each air line can be rolled as a unit from -90° to 90°.

A. High-Pressure Air Sting-Strut Model Supports.- The high-pressure air sting-strut model supports were designed for research on nozzle performance, afterbody/nozzle/empennage integration, and specific configuration development. There are two sting-strut supports available in which the model can be tested either at only 0° yaw or at 0° to 5° yaw (model yawed with respect to strut). The only difference between these two supports is the method of attachment of a high-pressure plenum/model support to the top of the strut. This sting-strut support system has been extensively used for research on isolated single-engine nozzles (fig. III-3), podded twin-engine nozzle performance and empennage integration (fig. III-4), afterbody/nozzle performance (partially metric model) of specific configurations (fig. III-5) and airplane/nozzle performance (entire model metric) of specific configurations (fig. III-6).

A sketch of the basic sting-strut support is presented in figure III-7 and a photograph showing specific parts of this support is presented in figure III-8. The sting-strut attaches to the main tunnel support at Tunnel Station 141.94 by use of a knuckle and inner sting (labeled sting "butt" on figure III-1). This arrangement will place the model center line at either 22 inches above or below the wind tunnel center line. The model can be located on the tunnel center line by use of the offset butt shown in Section II. The sting portion of the support system has a 2.0- by 6.0-inch rectangular cross section capped on top and bottom by half-cylinders of 1.0 inch radius. The strut is 5 percent thick with a 20.0 inch chord in the streamwise direction. The strut leading and trailing edges are swept 45°.

High-pressure air for exhaust simulation is routed through a 1.5 in. diameter gun drilled hole in the sting and six 0.625 in. diameter gun drilled holes in the strut. A flexible high-pressure (1800 psi) air hose connects the air supply from inside the strut head (fig. III-2) to the downstream end of the end of the sting. Instrumentation leads are routed through upper and lower passages in the sting and through passages in the strut leading and trailing edges.

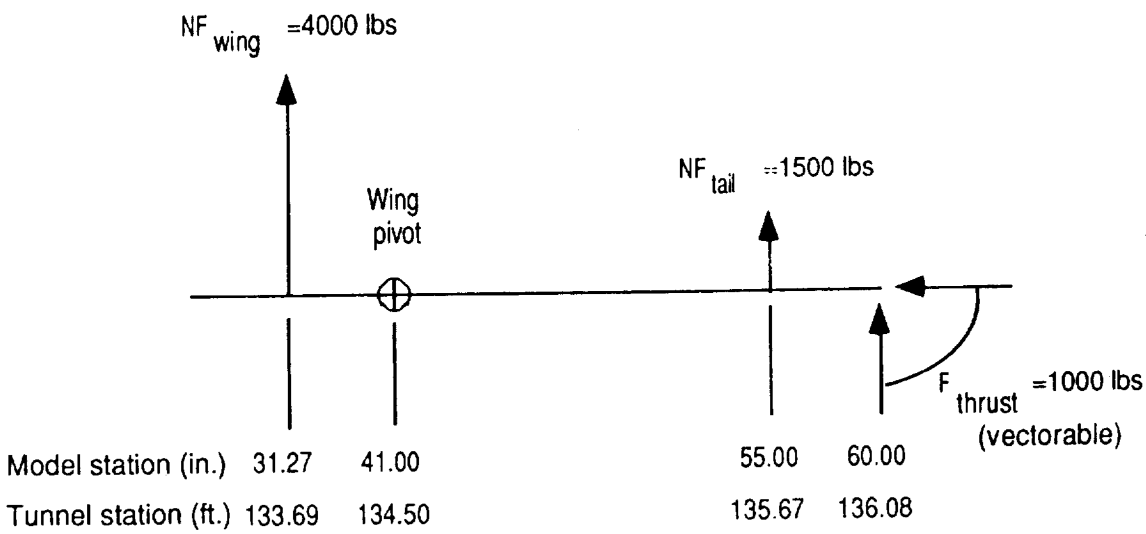
Model hardware is attached to the top of the strut, by use of a high-pressure plenum. The method of attachment of the high-pressure plenum to the top of the strut is different for the two available sting-struts. These differences are shown in figure III-7(a) for the 0° yaw sting-strut and in figure III-7(b) for the variable yaw sting-strut. Sketches showing the high-pressure plenums available for each of the sting-struts are presented in figure III-9. The air supply is then directed aft for single or multiple engine simulation. Equipment is available for simulating (and measuring) both single and twin-engine configurations. These propulsion simulation systems are described in Section IV. Design loads for the sting-strut support system acting through Tunnel Station 134 are given below:

Normal:	5,000 lbs.
Axial:	800 lbs.
Side:	500 lbs.
Pitch:	15,000 in.-lbs.

B. Wing-Tip Model Support.- The wing-tip model support was designed for basic research on afterbody/nozzle integration of twin-engine fighter aircraft. A sketch of this support system is presented in figure III-10 and photographs of the support system installed in the 16-Foot Transonic Tunnel are shown in figure III-11. High-pressure air is used for exhaust flow simulation.

The support system design philosophy was to provide a high angle-of-attack, high-load support system which also provided a realistic flow field for afterbody/nozzle research. The model support system attaches to the strut head tunnel support system through two pair of "V" struts (fig. III-10). The aft pair of struts are attached to the top of the tunnel support strut head and the forward struts attach through a bullet nose fairing to the forward end of the strut head at Tunnel Station 141.94. Air lines are routed through the forward struts and instrumentation lines are routed through the aft struts. Twin booms attach to the top of the "V" struts and carry air and instrumentation lines forward to the model. The boom leading edges were designed to minimize shock interactions with the model afterbody at $1.0 < M < 1.3$. An integral wing/centerbody is supported between the booms near Tunnel Station 134 (center of rotation) and routes air and instrumentation lines to the centerbody. All other model hardware attaches directly to the wing/centerbody. The uncambered wing planform was designed to be typical of current fighter type aircraft. Wing thickness ratio is realistic near the root, but because of strength and volume (to accommodate air and instrumentation lines) requirements, becomes progressively distorted near the tip. A sketch of the wing is provided in figure III.12. The wing centerbody is a constant area, 5- by 10-inch body with rounded (1.0 inch radius) corners. The existing forebody shown in figure III-10 represents a configuration with twin side inlets which are faired over for propulsion tests. This forebody is not a structural member of the support system and can be replaced with a different design if desired.

Design loads for the wing-tip support system are given below:



Note: Tunnel stations given for support with boom in most forward location.

This support system has several special features as listed below.

1) Rotating wing - The wing shown in figures III-10 to III-12 may be rotated with respect to the support booms. This feature allows a

large angle-of-attack range without bent knuckles. Hardware is available for the following wing rotations:

Wing incidence angle, deg.	Angle-of-attack range, deg.
0	-10 to 25
8	-2 to 33
16	6 to 41
32	22 to 57
60	50 to 85

2) Remote air-flow control valves - Electric flow control valves are located in the support bullet nose fairing. This feature allows balancing of left and right hand engines remotely from the control room.

3) Model position in tunnel - As shown in figure III-10, the model longitudinal position in the tunnel test section can be varied 6 or 12 inches by sliding the booms forward or aft on the support "V" struts.

4) Wing position - The wing shown in figure III-10 is in a high wing position. The wing support system can be inverted to provide a low wing type configuration.

In addition to the fighter-type wing shown in figure III-10, a straight and swept wing are available for nacelle/pylon integration studies as seen in figures III-13 and III-14.

C. <u>Boeing Sting Strut</u>.- The Boeing sting-strut was designed for research on a large-scale twin-engine supersonic-cruise fighter. A sketch of this sting-strut is presented in figure III-15. This sting-strut must be used with the strut head extender (fig. II-6). The sting has a 6.50 inch diameter and the strut has a streamwise chord of 20.00 in. The leading and trailing edges are swept 45°. The overall length of this support system is 88.44 in. There are no Langley balances that can be used directly with this sting-strut.

This sting-strut can be used with both a single or dual air supply. High-pressure air for exhaust simulation is routed through 1.5 in. diameter holes at the end of the sting to 1.25 in. diameter tubing to the strut. Connection to the model air lines is made at the bottom on each side of the strut. (See fig. III-15.) Design loads for this sting-strut acting through Tunnel Station 134 are as follows;

Normal:	4,800 lbs.
Axial:	360 lbs.
Side:	480 lbs
Pitch:	2,100 in.-lbs.
Yaw:	1,680 in.-lbs.
Roll:	1,680 in.-lbs

D. Air-Sting Support.- The air-sting support was designed for research on a subsonic transport configuration with twin engines (one engine on each wing). Information on the propulsion simulator system (internal hardware) designed to be compatible with this support can be found in Section IV of this document. The flow-through 1627 balance/bellows assembly was designed for use with this support and additional information on this balance is contained in Section VI of this document. This air-sting support with either the 1627 balance/bellows assembly or other internal hardware could be used for research on other type model configurations. A sketch of the air-sting support is presented in figure III-16.

The air-sting is attached directly to the main tunnel support at Tunnel Station 141.94 and consequently does not use a knuckle or sting butt (figure III-1). The sting forward of the taper has a 2.75- by 2.25 in. rectangular cross section capped on top and bottom by half-cylinders on 1.375 in. radius.

Two 1.75 inch diameter gun drilled holes are in the sting. High-pressure air for exhaust simulation is supplied from the fixed air supply within the strut head and is routed through the upper sting passage. Instrumentation leads are routed through the sting lower passage and two 0.75 inch diameter holes drilled through near the sting-model attachment point. Model hardware is attached to the upstream end of the sting by eleven 1/2-20 NF-3 bolts.

Design loads for the air-sting acting through Tunnel Station 134 are given below:

Normal:	3,000 lbs.
Axial:	500 lbs.
Side:	1,000 lbs.
Pitch:	15,000 in.-lbs.
Yaw:	10,000 in.-lbs.
Roll:	10,000 in.-lbs.

E. <u>LTV Air-Sting Support</u>.- The LTV air-sting has been used with several model tests in the 16-Foot Transonic Tunnel. A sketch of this air-sting, which has a circular cross section with 4.25 inch diameter, is provided in figure III-17. The original test utilizing this support was the

ADAM-II model and the support was later used with a General Dynamics VEO-wing model.

The downstream end of the LTV air-sting fits the 16-Foot Transonic Tunnel butts. Because of the male taper on the downstream end, this sting cannot be used with most 16-Foot Transonic Tunnel knuckles; however, a 5° and 12.5° adapters are available to either increase angle-of-attack range or to provide sideslip capability. Use of these adapters increases total support length. The upstream end of the LTV air-sting was designed to accept the VTB-3 flow-through air balance. Special adapters can be designed and built in order to use other balance designs.

The LTV air-sting is generally mounted in the wind tunnel such that the sting is on the tunnel center line. High-pressure air is routed to the model through a 2.0 inch diameter gun drilled hole in the sting. All instrumentation leads are routed external to the sting and taped in place when the sting is used with propulsion models. Unless provisions were made to route the instrumentation leads through the sting "butt", a model using this support system could not be remotely rolled at the strut head. By use of an adapter (not currently in existence) at the balance taper end, this sting could also be utilized for aerodynamic model tests with the instrumentation leads being routed internal to the sting.

F. Alternate Wing-Tip Support.- An alternate wing-tip support such as used with an F-18 powered model can be installed in the 16-Foot Transonic Tunnel with the air sting shown in figure III-18 and the adapter butt shown in figure II-11. A pair of struts attach to this sting and booms to support the model exist but are not kept at the 16-Foot Transonic Tunnel.

G. Semi-Span Support Assembly.- A sketch of the 16-Foot Transonic Tunnel semi-span support assembly is shown in figure III-19. This support system consists of a 13- by 6-foot reflection plane located about 5.2 feet from the lower test section wall or about 2.5 feet from the wind tunnel center line. The model axis of rotation is located at Tunnel Station 134. High-pressure air is routed to the model by use a of flexible air line at airflow rates of up to 15 lbs/sec. This support system was designed to support a model weighing 500 lbs. having the following aerodynamic loads:

Normal:	5,000 lbs.
Axial:	250 lbs.
Side:	0 lbs.
Pitch:	12,000 in.-lbs.
Yaw:	5,000 in.-lbs.
Roll:	105,000 in.-lbs.

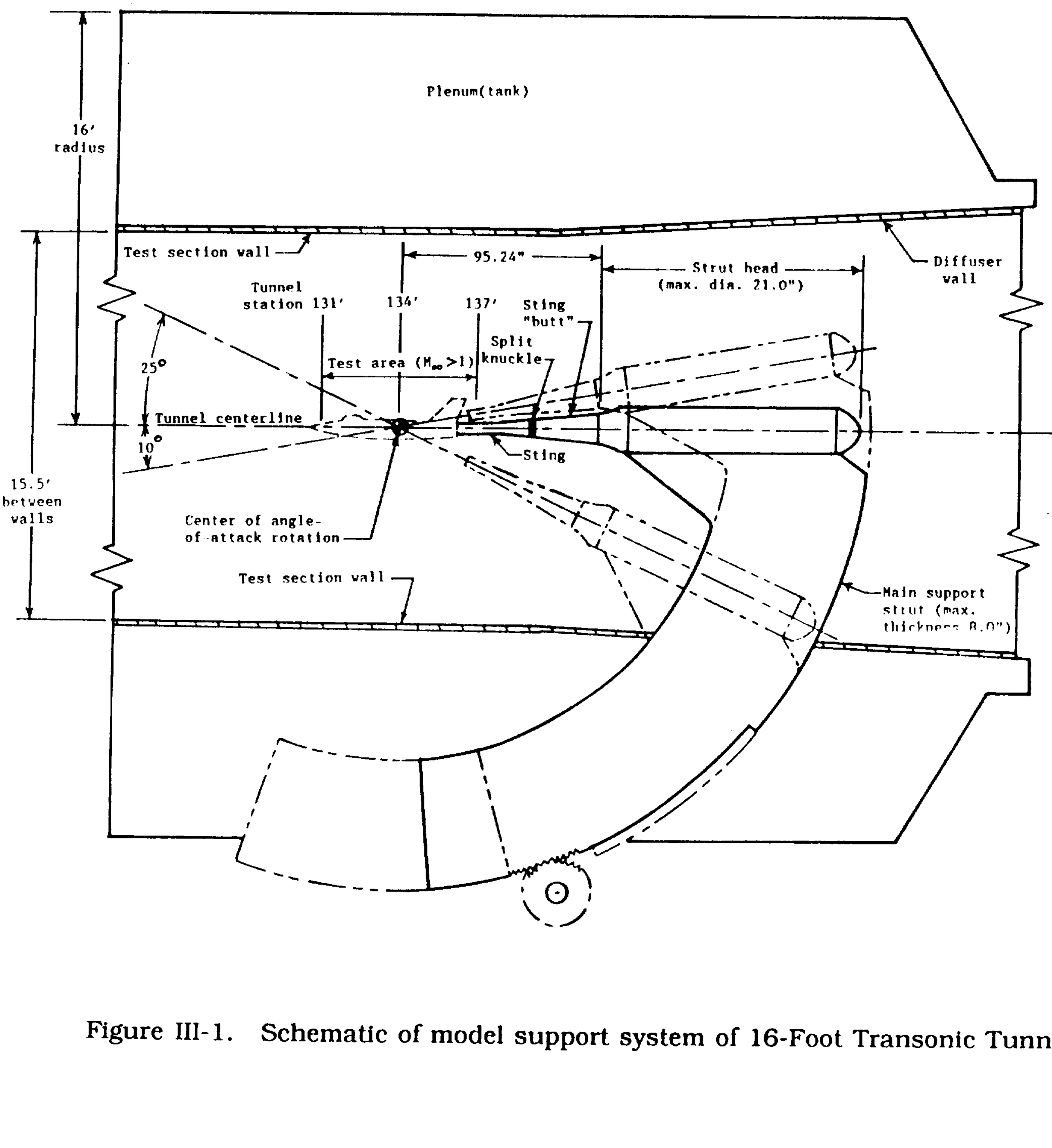


Figure III-1. Schematic of model support system of 16-Foot Transonic Tunnel.

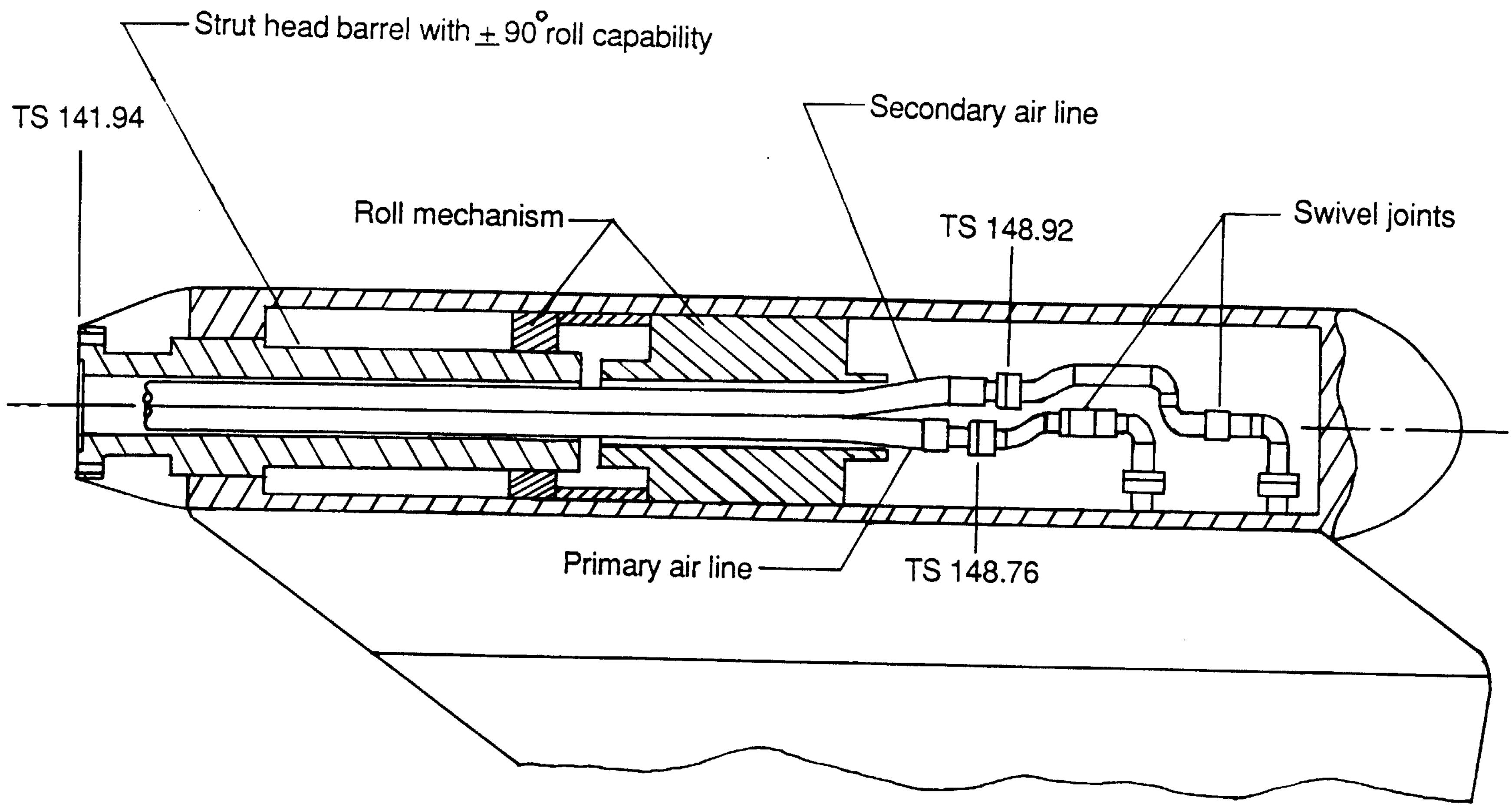


Figure III-2. **Sketch of tunnel support system strut head.**

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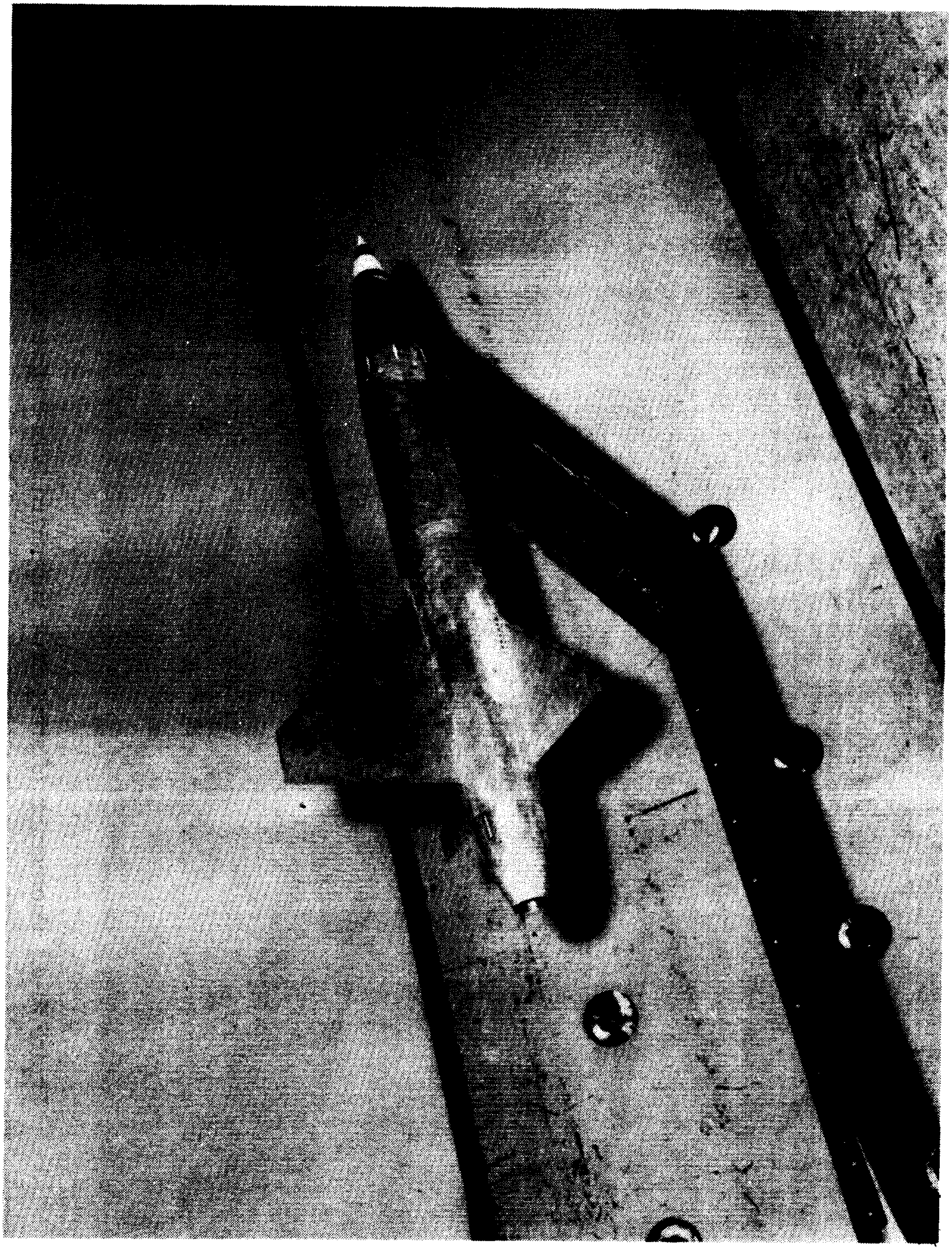

Figure III-3. Photograph of sting-strut supported single-engine configuration.

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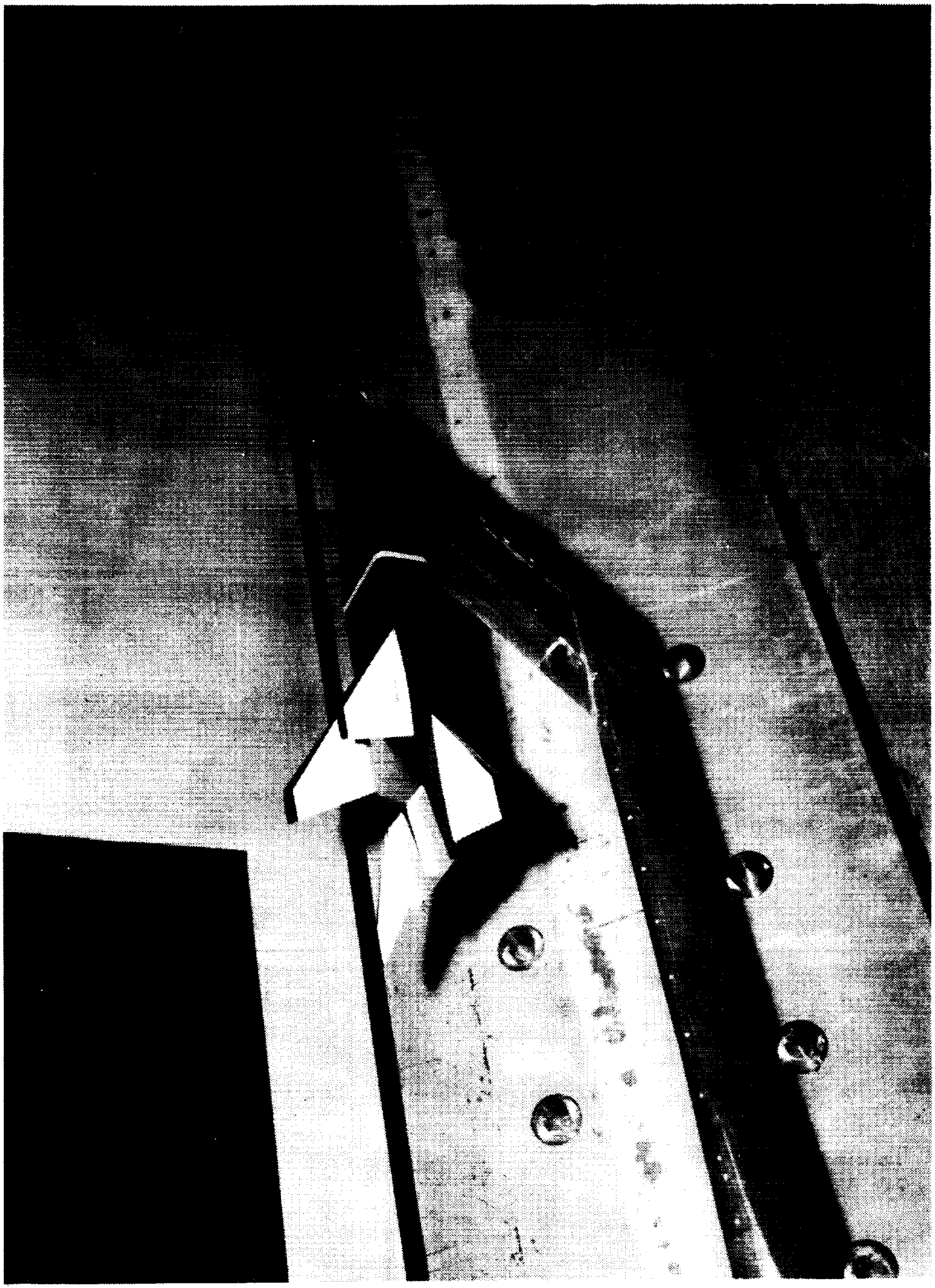

Figure III-4. Photograph of sting-strut supported twin-engine configuration.

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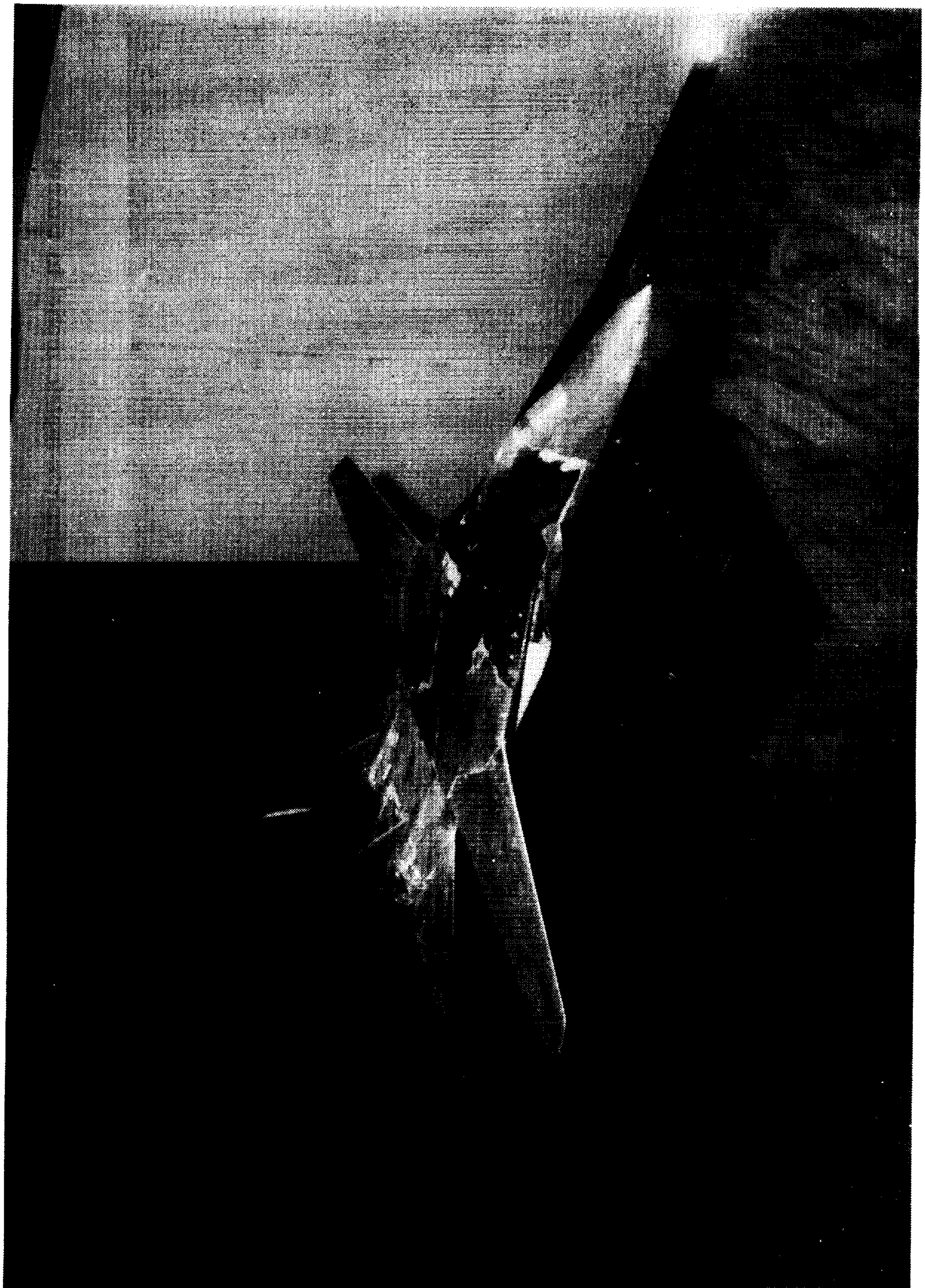

Figure III-5. Photograph of sting-strut supported partially-metric model in tunnel.

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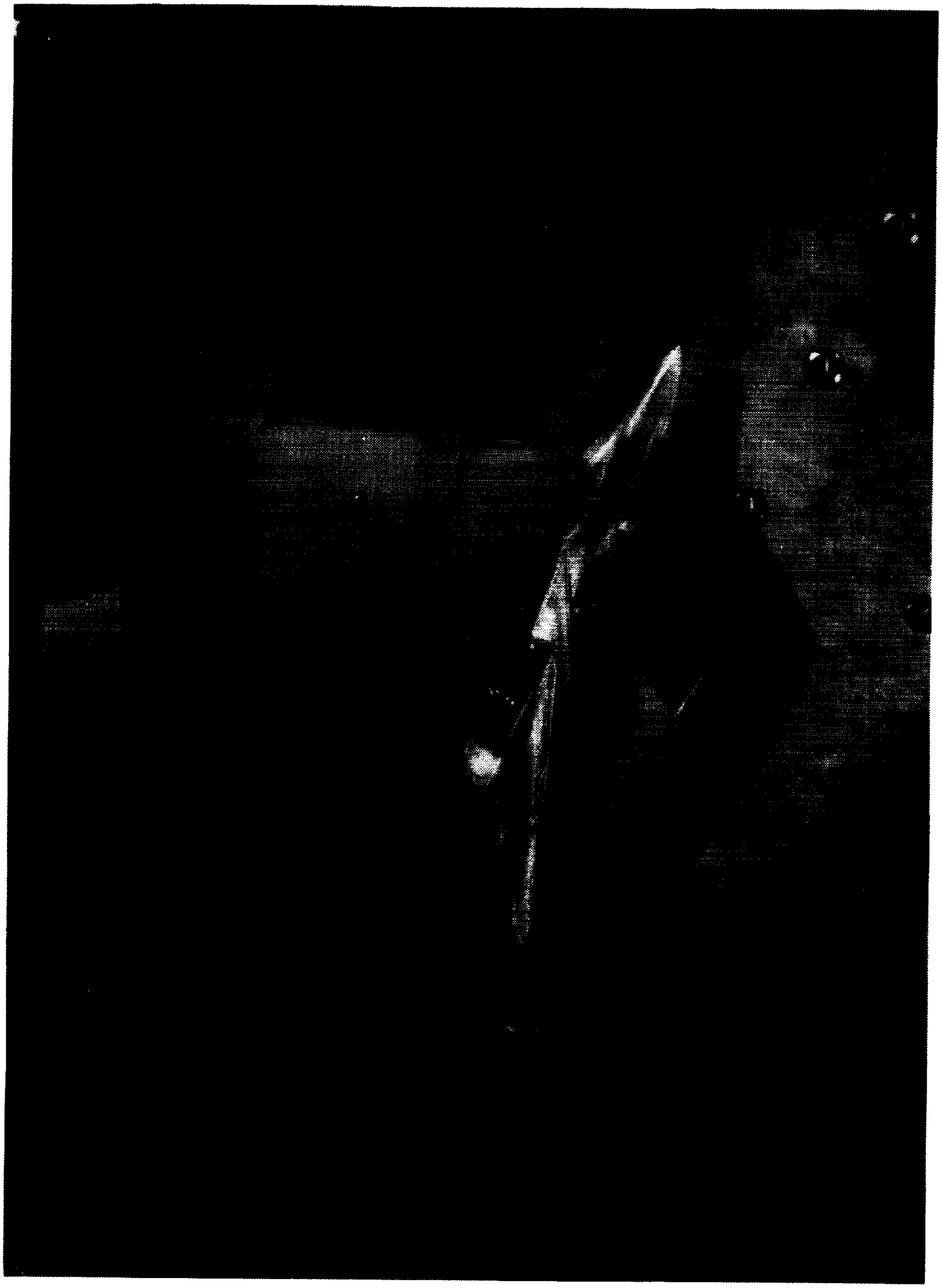
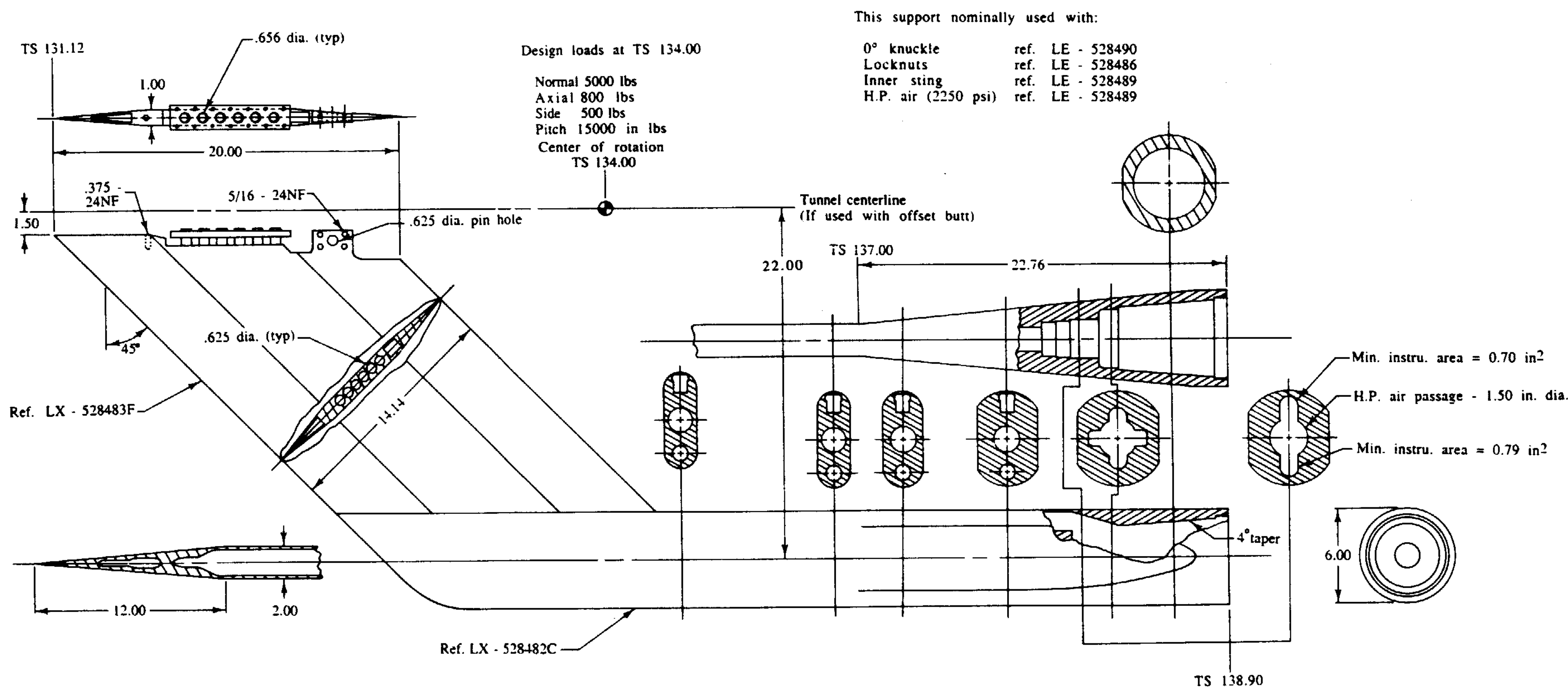
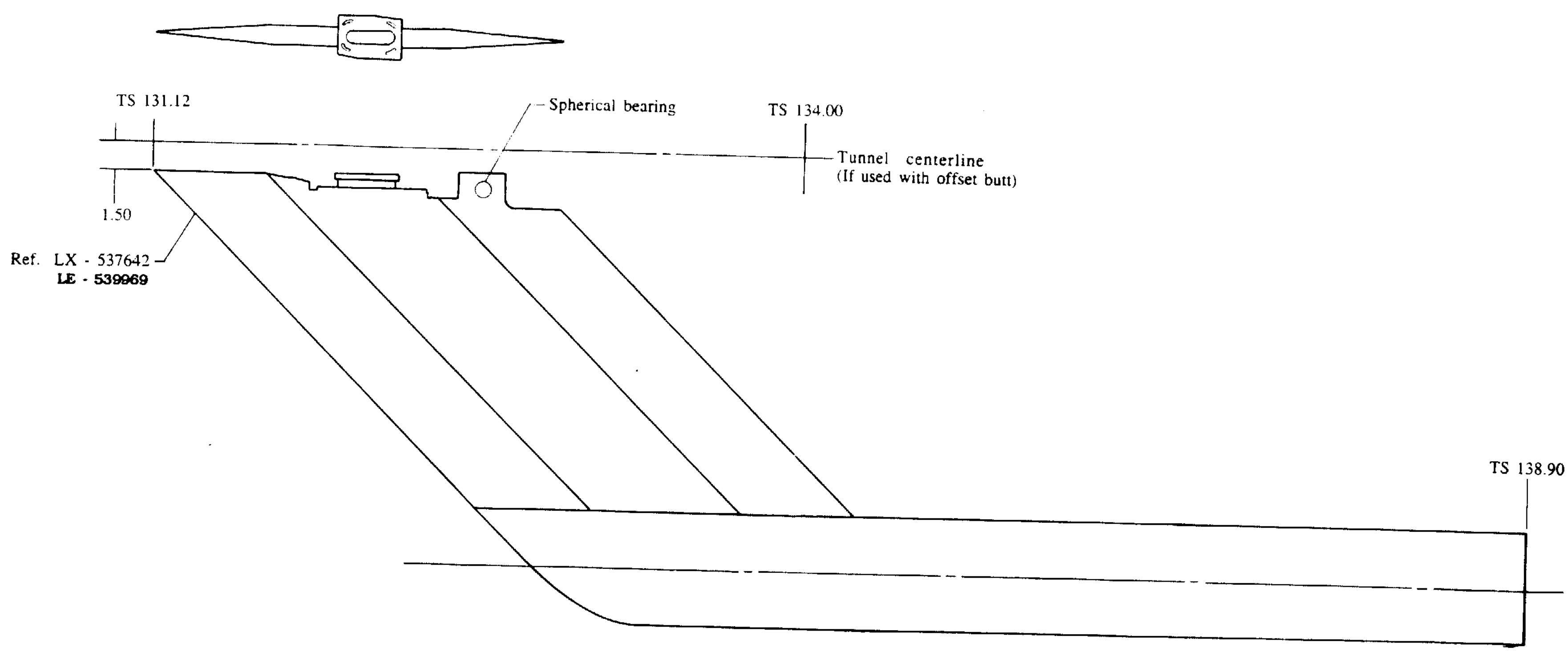


Figure III-6. Photograph of sting-strut supported complete-metric model in tunnel.



(a) No yaw

Figure III-7. Sketch of high pressure air sting-strut support system. All tunnel stations are in feet. LX-528482C.



(b) Yaw strut

Figure III-7. Concluded.

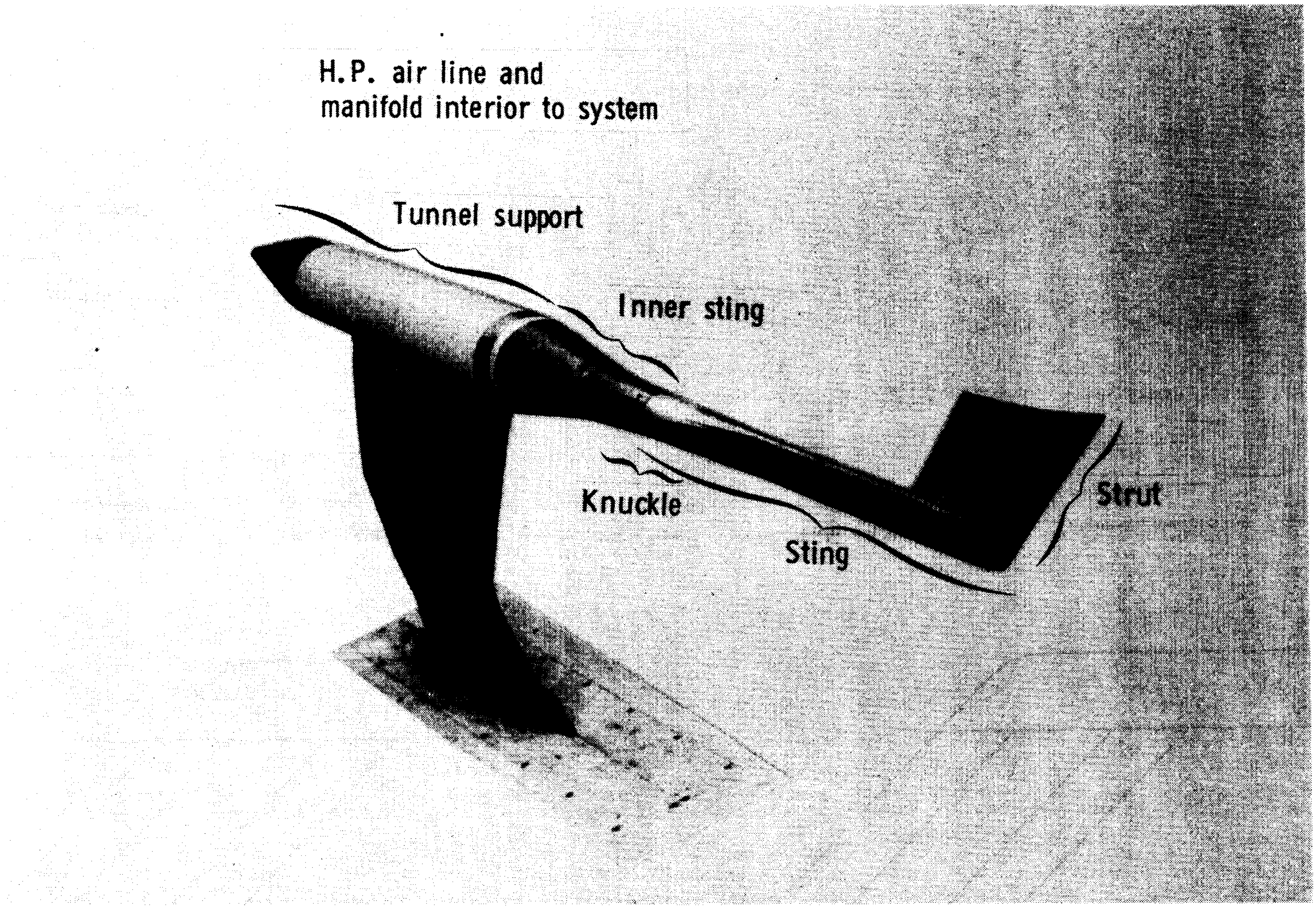


Figure III-8. Photograph showing parts of sting-strut support system.

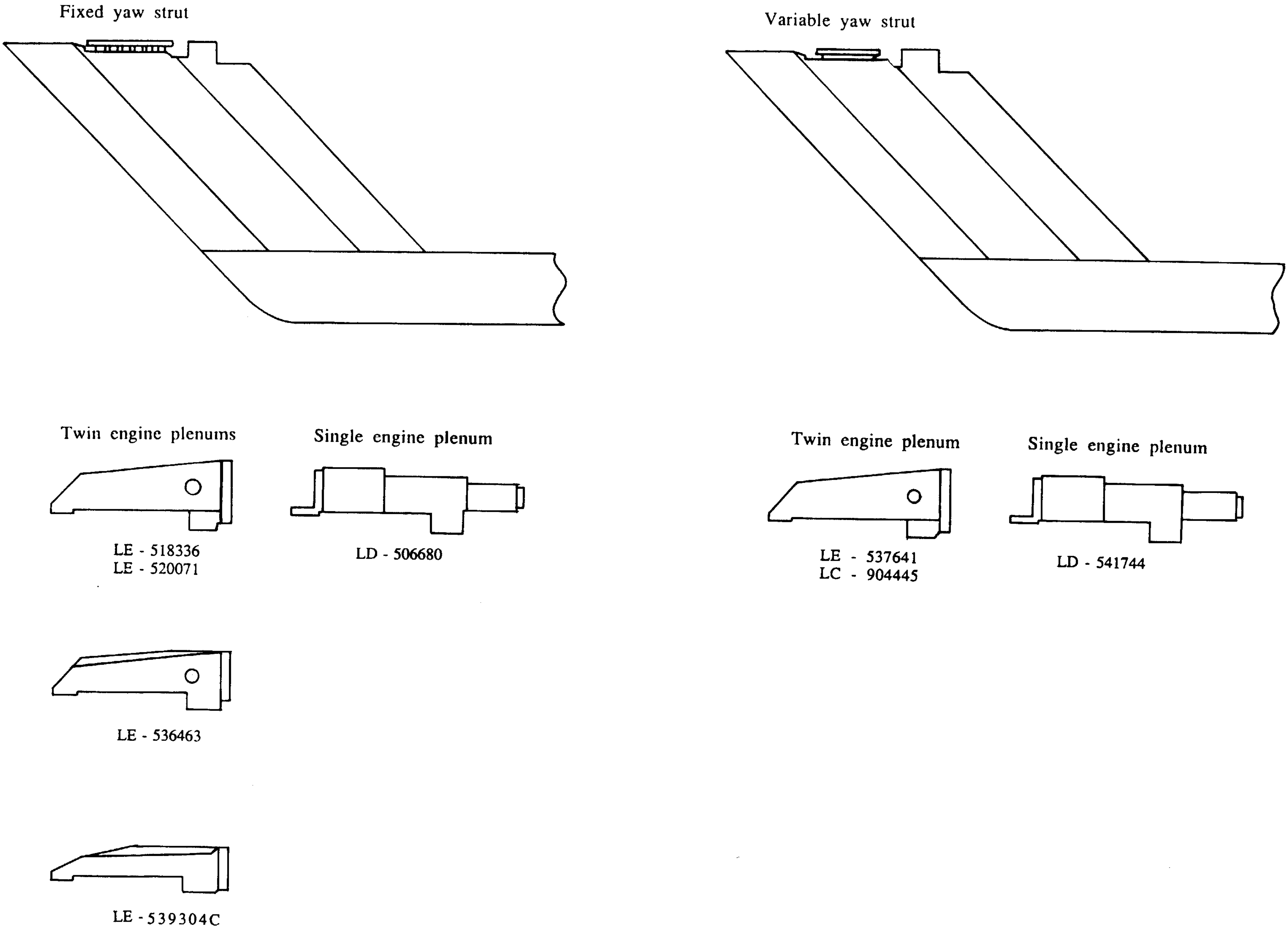


Figure III-9. Sketch showing available high pressure plenum.

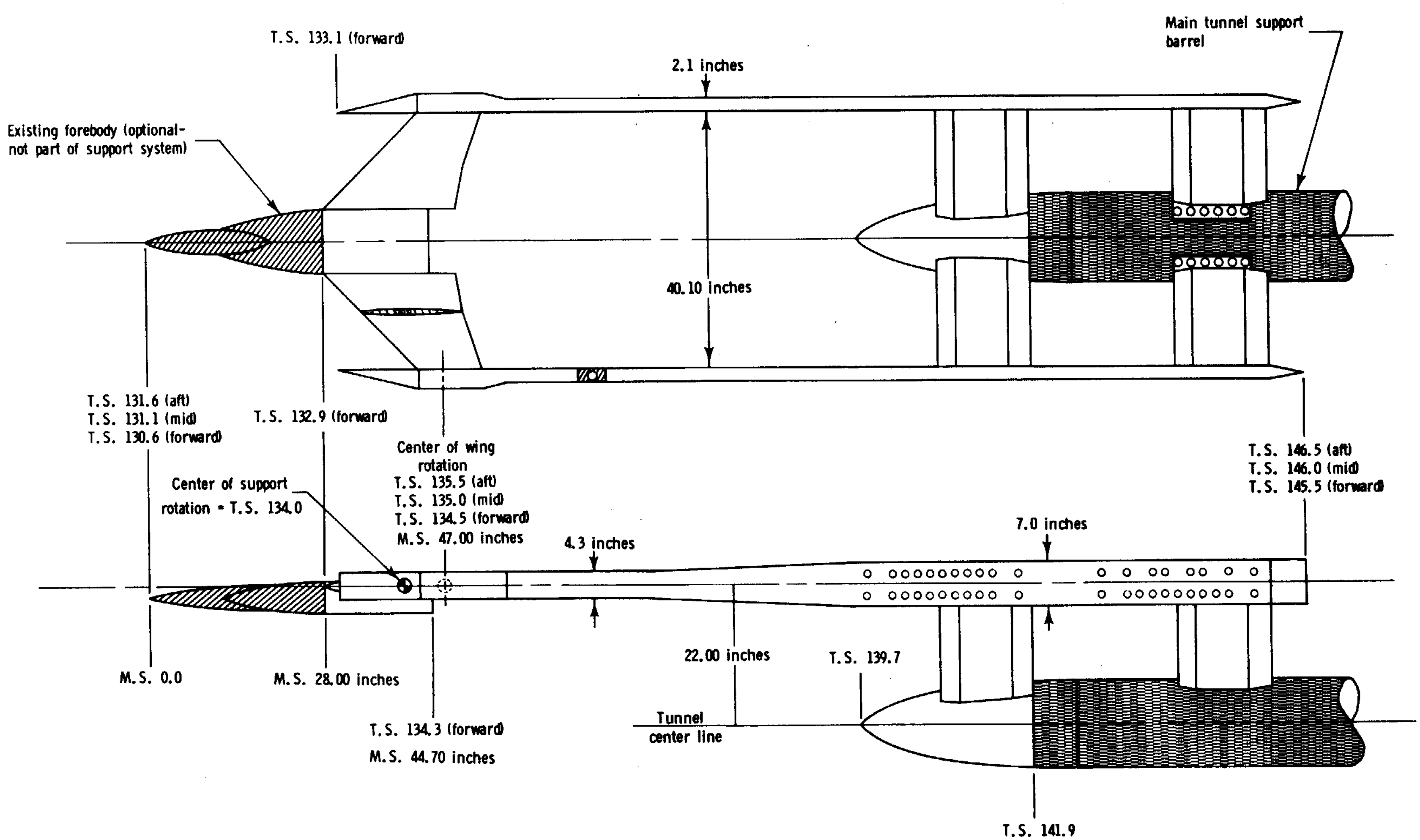


Figure III-10. Sketch of wing-tip support system. All tunnel stations are in feet. All model stations (MS) are in inches. Booms are shown in forward location. Wing/centerbody shown in high wing position.

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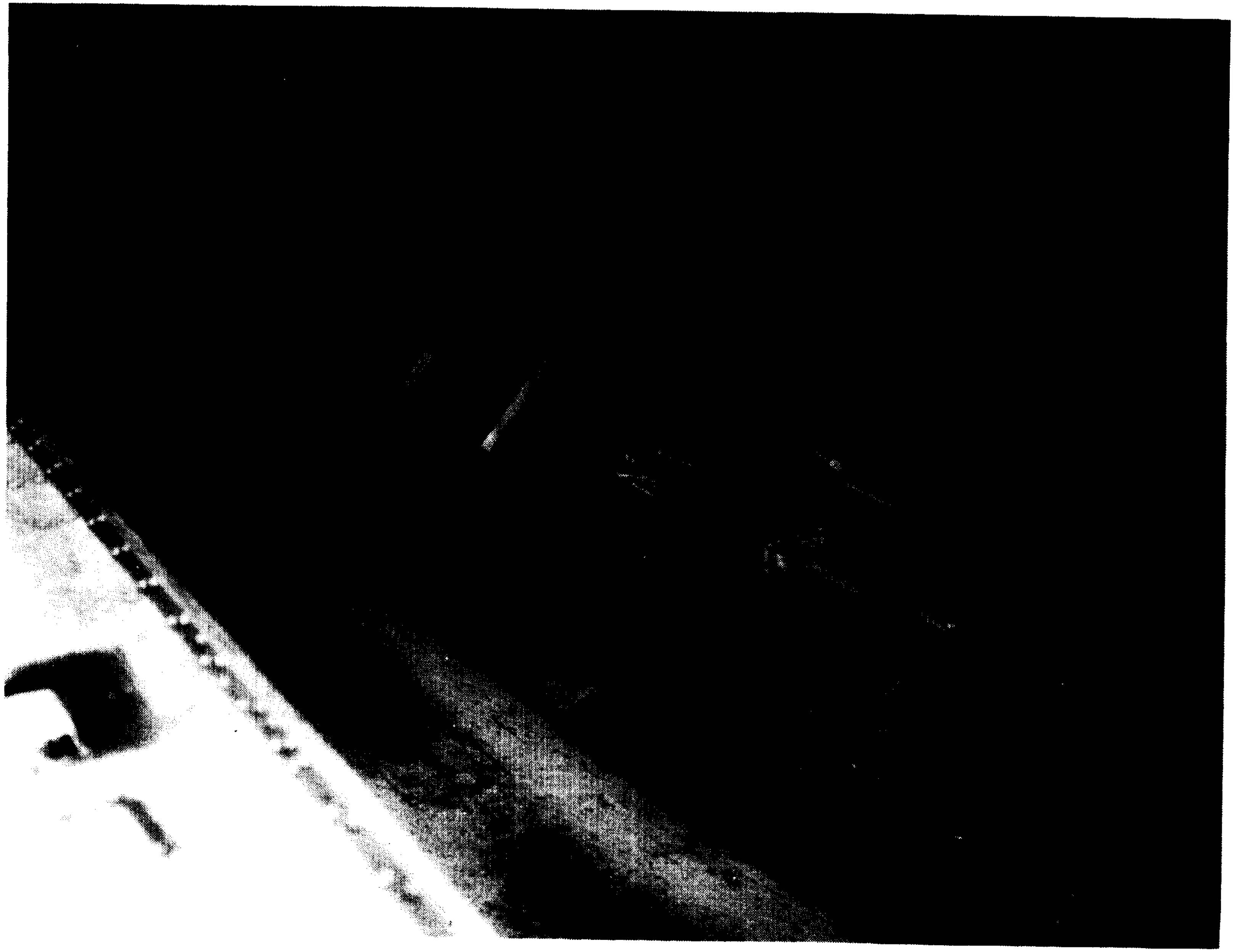

Figure III-11. Photograph of wing-tip support system.

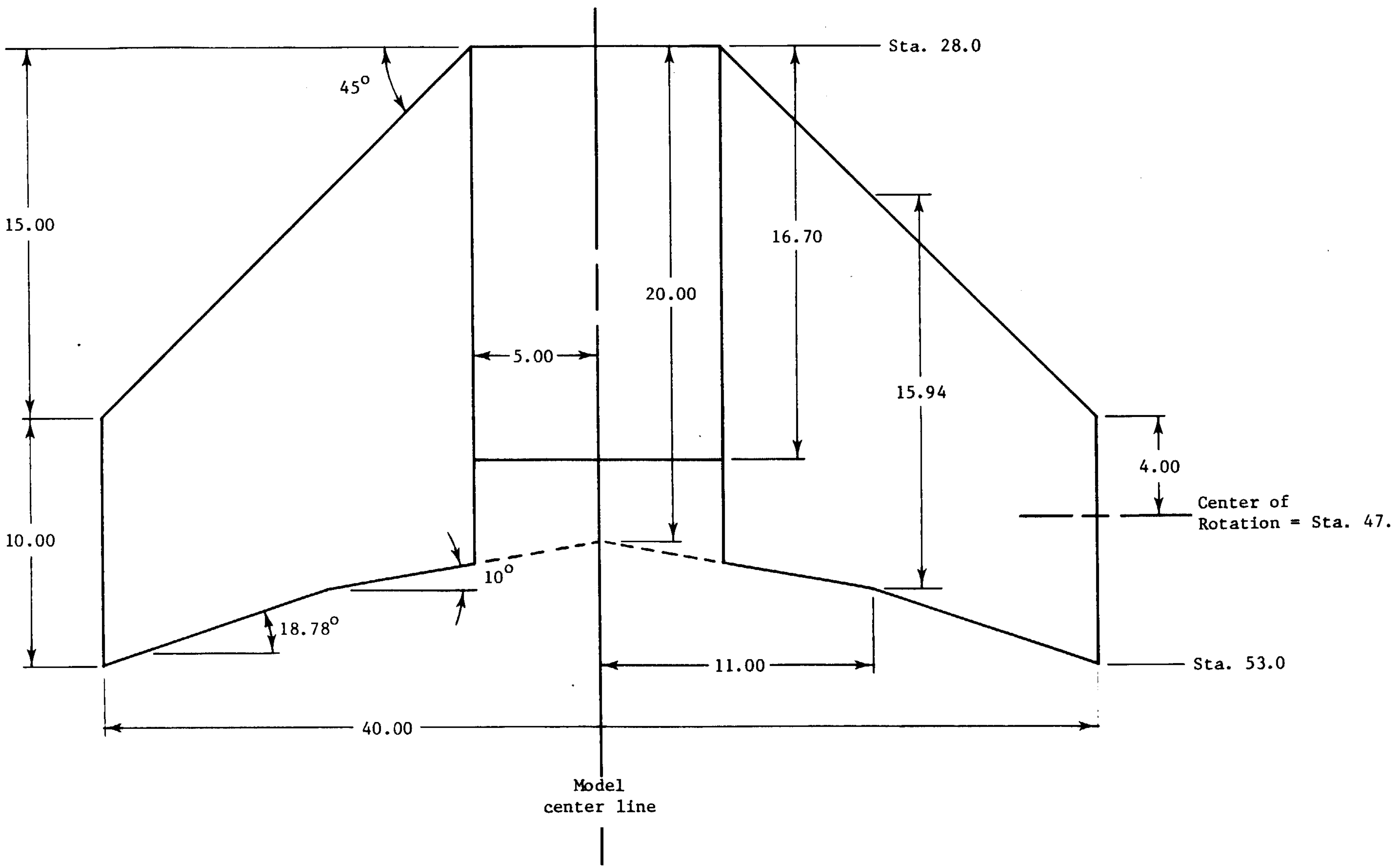

Figure III-12. Wing details for wing-tip support system. (All dimensions are in inches.)

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Figure III-13. Photograph of straight wing configuration.

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Figure III-14. Side view of model with under-the-wing nacelle and clockwise rotation propeller.

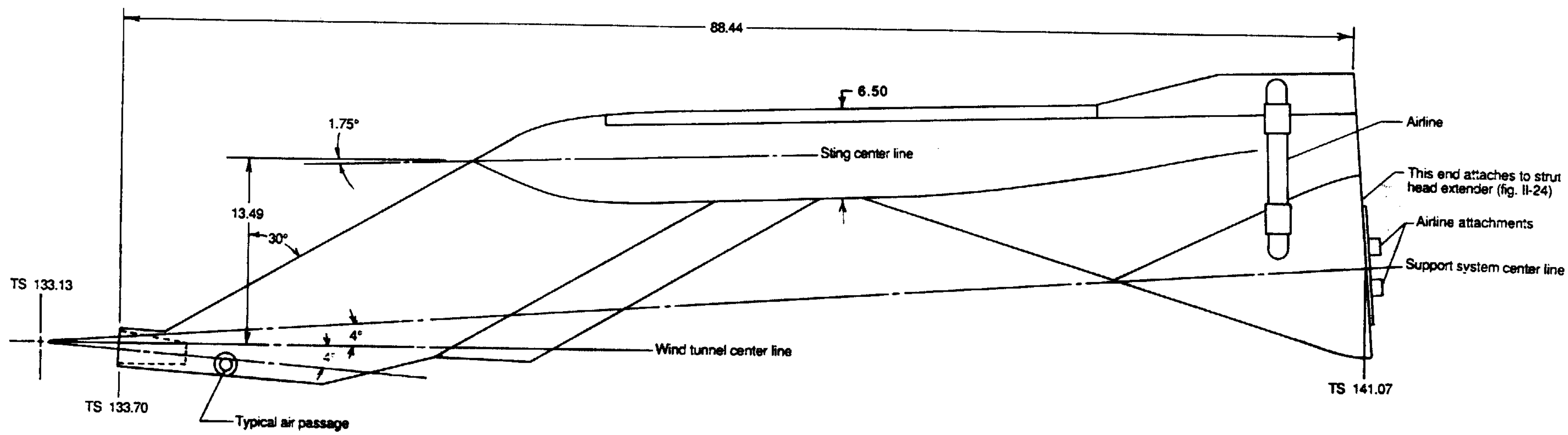

Figure III-15. Sketch of Boeing sting-strut support system. All tunnel stations are in feet.

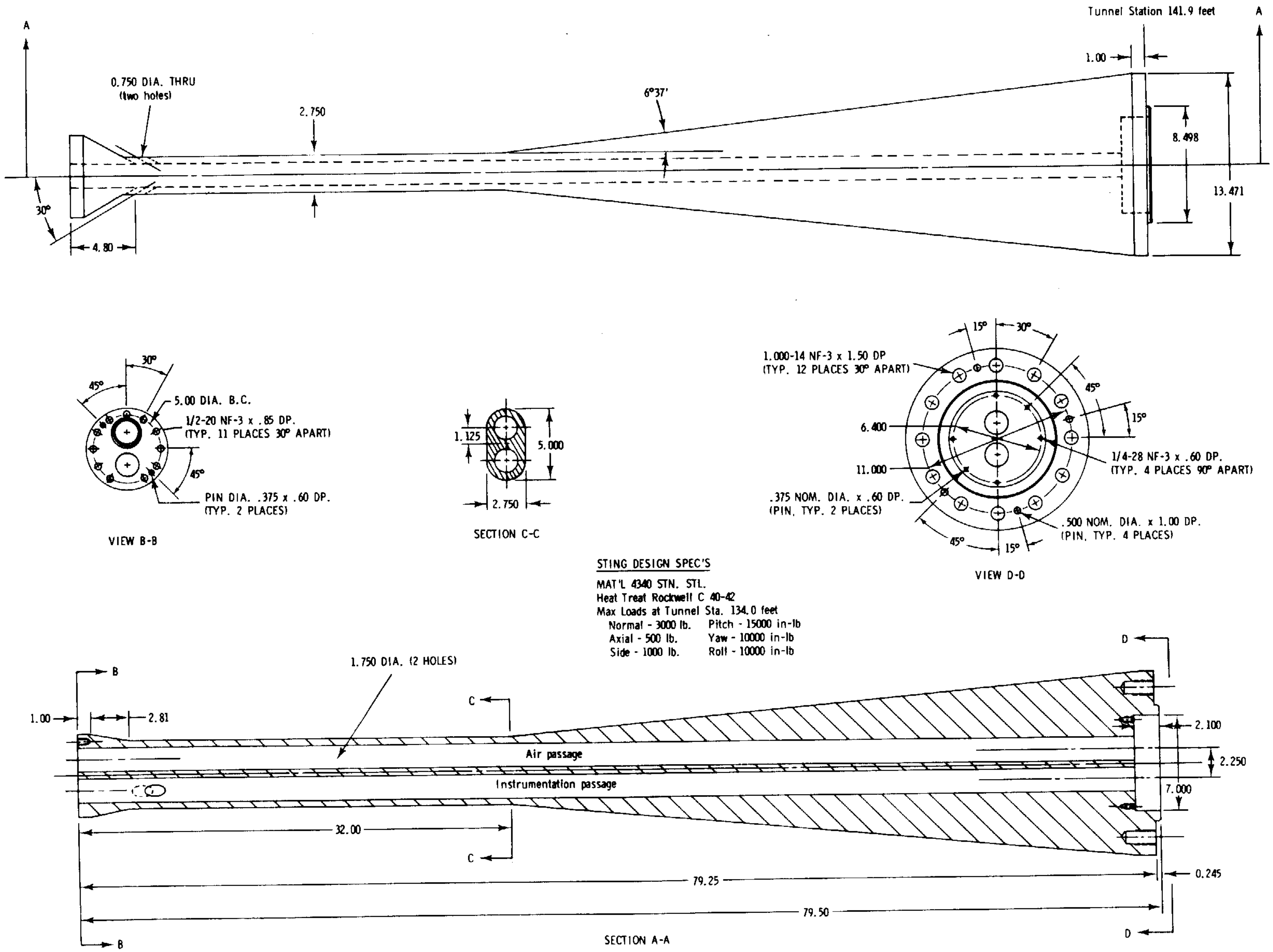

Figure III-16. Sketch of air sting. (All dimensions are in inches unless otherwise indicated.) Ref. Dwg. LE-534599.

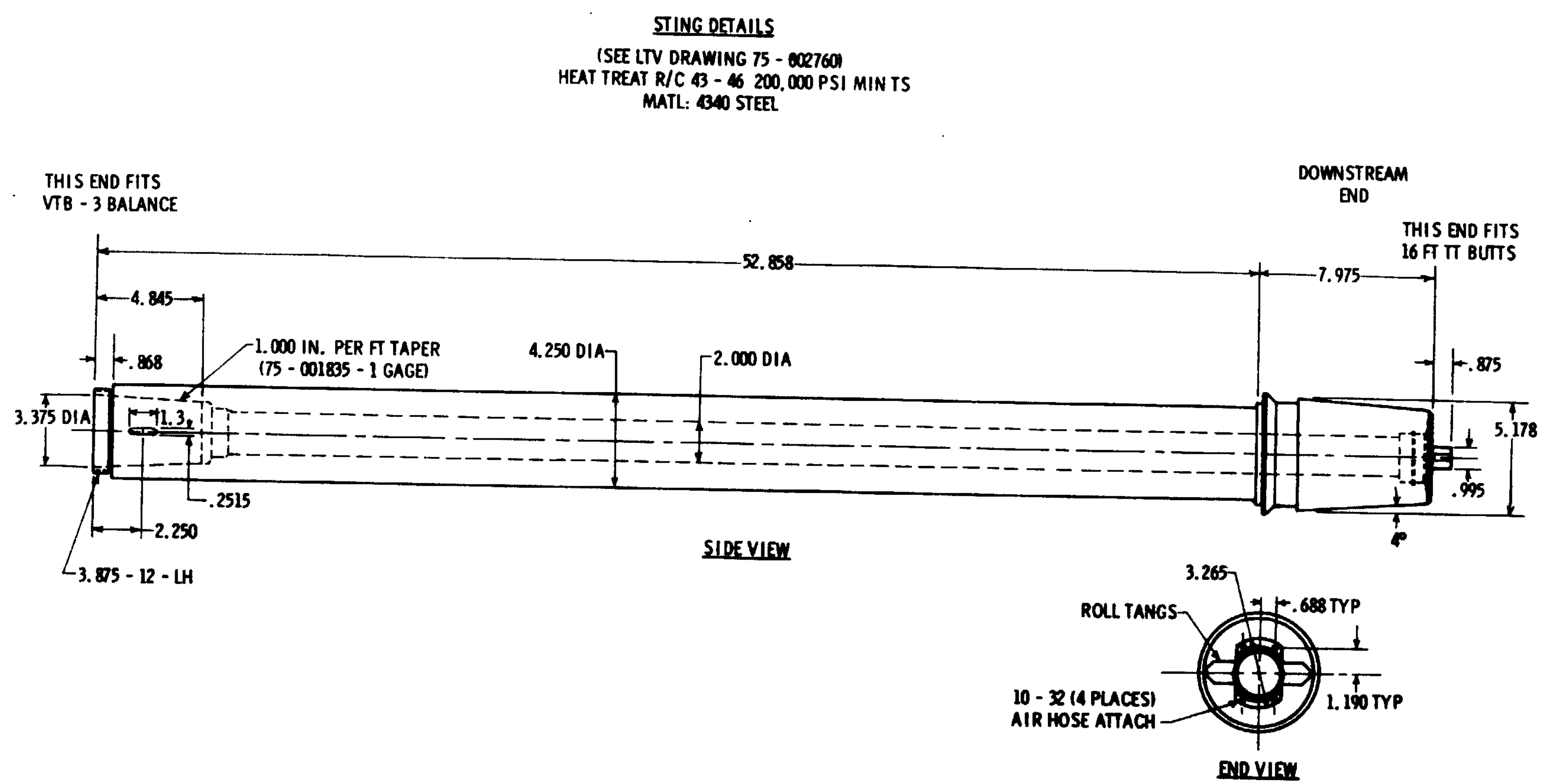

Figure III-17. Sketch of air-sting support. (All dimensions are in inches.) Ref. Dwg. LTV 75-002760.

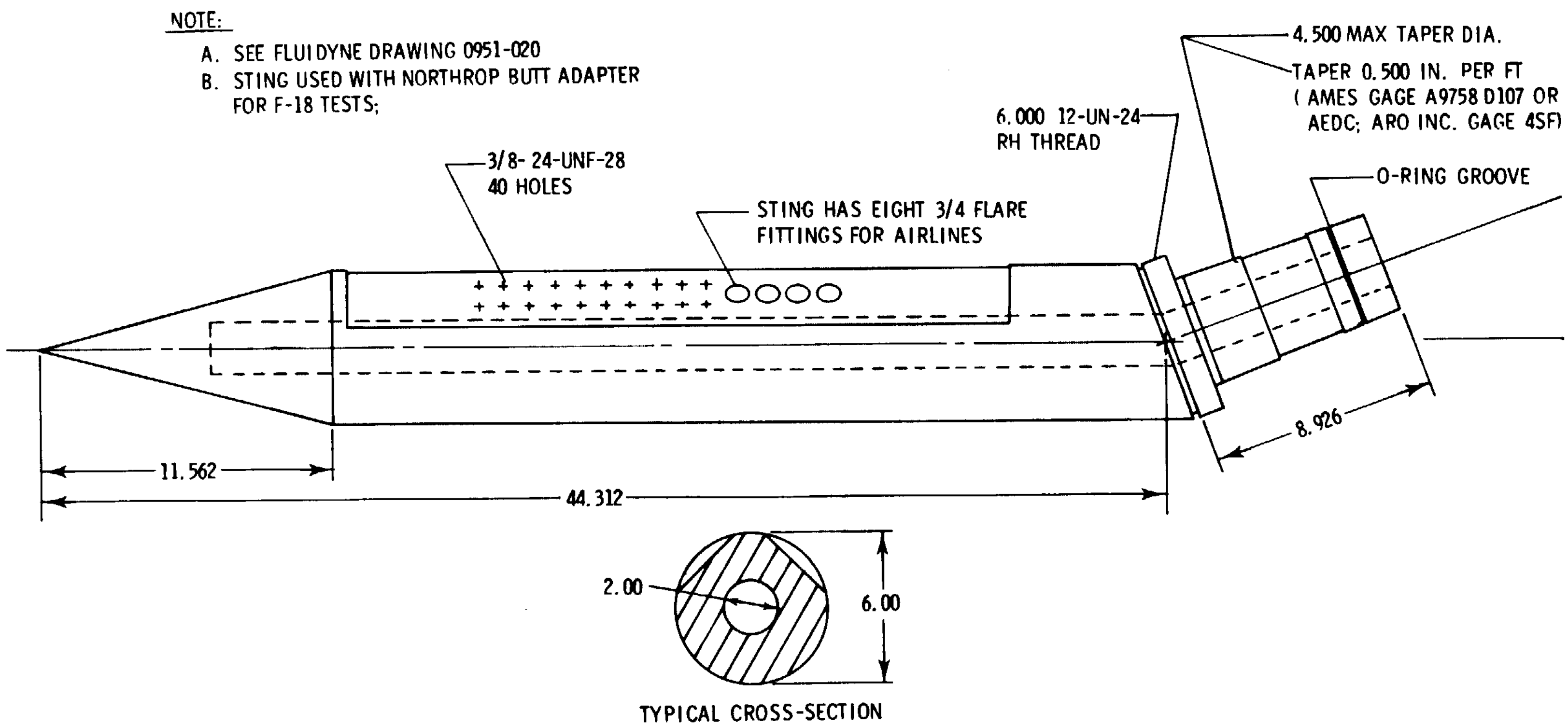

Figure III-18. Sketch of Northrop air sting 0951-020. (All dimensions are in inches.)

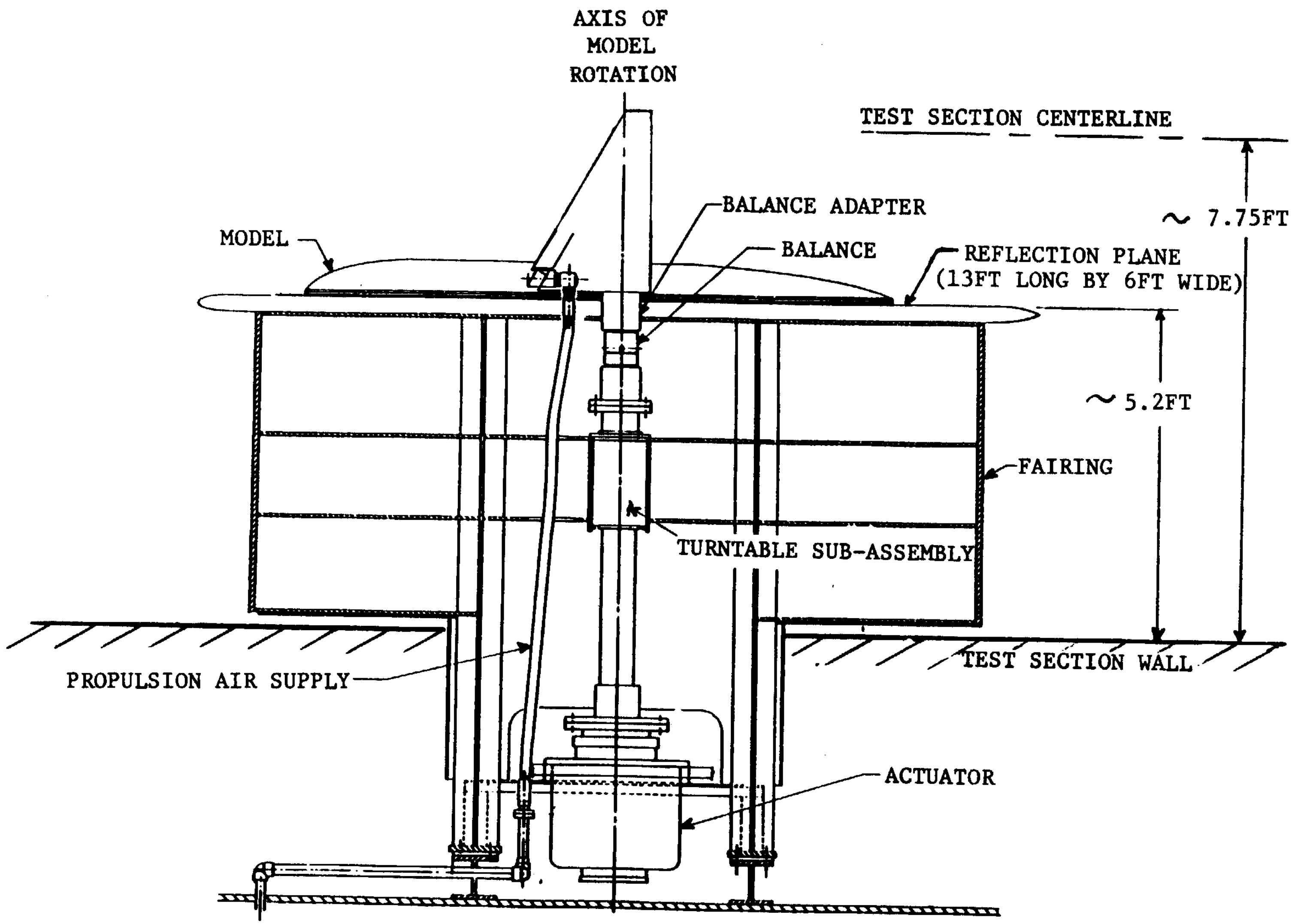

Figure III-19. Sketch of semi-span model installation with propulsion system air lines.

SECTION IV - Propulsion Simulation Systems

A. Single-Engine Propulsion Simulator Systems.- The single-engine propulsion simulators were designed primarily for research involving installed isolated nozzle performance. The five simulator systems (axisymmetric and nonaxisymmetric) shown in figures IV-1 to IV-5 are installed on the sting-strut model support discussed previously in the support system section. Except for external shape, all systems are similar in that high-pressure air is ducted through the sting strut into the high-pressure plenum and into a single bellows arrangement where the transfer of high-pressure air from nonmetric to metric portions of the model occurs. The flow then passes through a choke plate, a flow straightener (or a combination of the two) into an instrumentation duct which typically contains a total pressure rake and a total temperature probe. Forces and moments on the metric portions of the model are measured by strain gage balances (NASA 1614, 1631, etc.). Maximum flexibility was designed into the systems with respect to balance selection and possible hardware variations.

The axisymmetric single-engine simulators shown in figures IV-1 and IV-2 are essentially bodies of revolution with semi-ogive noses. Maximum diameter of the simulators is 6.0 and 7.34 inches, respectfully. The metric break is located 26.5 inches aft of the nose on the 6.0 inch system and 42.128 inches aft of the nose on the 7.34 inch system. The metric portion of the model containing the bellows arrangement begins just aft of the metric break. The instrumentation section attaches directly to the low-pressure plenum which contains the bellows section. Hardware aft of the instrumentation section depends upon test requirements and objectives. It should be noted that an additional nose (or forebody) exists for the system shown in figure IV-1, which is conical in shape and begins at station 6.0. The cone half angle remains at 14° just as the longer nose shown in figure IV-1; however, the radius that fairs the conical section to cylindrical section is considerably smaller.

The nonaxisymmetric single-engine simulators are shown in figures IV-3 to IV-5. The first of these (shown in figure IV-3) has a metric centerbody which extends from model station 26.5 to model station 48.4. This centerbody has a constant cross-sectional area, super-elliptical external shape which is defined by the equation:

$$\left(\frac{z}{3.1}\right)^9 + \left(\frac{y}{3.4}\right)^9 = 1$$

The nonmetric forebody provides a smooth transition from the conical nose (cone half angle of 14°) which ends at station 6.0 to the super-ellipse constant cross-sectional area of the model beginning at station 26.5. The bellows section and the instrumentation section end 40.95 and 48.4 inches aft of the nose, respectively. For the test configuration shown in figure IV-3 (reference IV-1), the nozzle is attached to the centerbody at station 48.4. The nozzle internal shape provides a constant cross-sectional area transition from circular (at station 48.4) to a super-elliptical shape. Hardware now exists which can be attached directly to the instrumentation section which provides a transition from circular to nonaxisymmetric (square with rounded corners) internal shape. This section is 6.65 inches long and terminates at station 55.05. The external geometry of this section is a constant cross section with the same dimensions as the centerbody. An additional constant cross-sectional area spacer is available which when placed between the instrumentation section and the transition section moves the nozzle connect station downstream by 9.36 inches.

The second and third nonaxisymmetric single-engine propulsion simulator systems are "sister" configurations. They were built with the maximum commonality in parts. Most of the attachment and internal propulsion simulator hardware are common. The two configurations are the same height, but differ in body width. Body number 1 (figure IV-4) is 6.0 inches high and 7.252 inches wide at the maximum (and constant area) cross section. The cross-sectional shape is rectangular between fuselage stations 27.00 and 53.00 with corner radii of 1.14. Body number 2, shown in figure IV-5, has a different ogive nose and is 10.126 inches wide at the maximum cross section. The corner radii are 1.05 inches and, therefore, slightly smaller than those of the narrow body configuration. The wide body configuration is assembled by simply bolting scab fairings to the narrow body low-pressure plenum.

B. Twin-Engine Propulsion Simulator Systems.- Three separate twin-engine propulsion simulators are currently available to 16-Foot Transonic Tunnel personnel: the single bellows twin-engine system, the sting-strut twin jet system, and the wing-tip-mounted twin jet system. As was the case with the single-engine simulators, the design concepts allow maximum flexibility with regard to balance selection and hardware arrangements.

The earliest twin-engine propulsion system, the single bellows twin-engine system (figure IV-6), was essentially built around the single-engine system discussed previously. As seen in the figure, much of the hardware including the sting-strut support, the high-pressure plenum chamber, and the flexible metal bellows is the same hardware found in the single-engine simulator systems.

High-pressure air is exhausted radially through the sonic nozzles into the low-pressure plenum chamber where it is separated into two flows by the two tailpipes which feed the twin nozzles. The main balance measures total forces and moments on the afterbody and nozzles and is the same class of strain-gage balance used in the single-engine systems. In addition, an afterbody drag balance (the NASA 1618) was used to isolate the afterbody drag from the total afterbody thrust-minus-drag. Again, however, the sketch in figure IV-6 (reference IV-2) is only one example of many possible test hardware arrangements.

The forebody fairs the conical-shaped nose (with a half angle of 14°) into the constant cross-sectional rectangular shape which is a 5-by 9-inch rectangular shape with 1 inch radii at the corners. This forebody is 33.0 inches long from the nose to the metric break (or afterbody connect station).

A sketch of the sting-strut twin jet system is shown in figure IV-7 and photographs of the internal hardware are shown in figure IV-8. As was the case with the single-engine simulator and the single bellows twin-engine system, high-pressure air is ducted through the sting-strut support system into a high-pressure chamber mounted on top of the sting-strut support system. As seen in figure IV-7 and reference IV-3, the high-pressure air is split into two separate flows and is ducted through the supply pipes to two separate flexible metal bellows assemblies. These separate air flows can be balanced by use of the manually operated flow control valves located in each air supply pipe. Once the air is transferred across the balance (by ducting through the bellows assemblies), it is exhausted out through the tailpipes feeding the nozzles. Total forces and moments on the afterbody (including the thrust components) are measured on the main force balance. In addition, a second balance is used to measure internal and external nozzle forces (thrust-minus-drag). The main force balance used in this example is the NASA 1621 and the thrust balance is the NASA 711. The constant cross-sectional area starting at FS 19.5 extends to FS 39.0 and is a 5-by 9-inch rectangular shape with rounded corners. The metric break (which is also the afterbody connect location) is located at FS 39.0. One variation of this system can be found in reference IV-4 where a 12.0 inch section was removed from the air supply pipes thus moving the metric break forward 12.0 inches.

Sketches of a typical wing-tip-mounted twin jet simulator test arrangement (reference IV-5) and details of the two available internal propulsion hardware systems (high loads system and low loads system) are presented in figures IV-9, 10, and 11, respectively. As indicated in figure IV-9, high-pressure air is ducted through the wing-tip support system centerbody into the two high-pressure plenums located on each side of the fuselage. These plenums direct high-pressure air into two separate flexible metal bellows assemblies where transfer of the air across the balance takes place. These twin bellows assemblies in turn supply the exhaust nozzles.

The high loads system (figure IV-10) was designed around the NASA 1617 strain gage balance. As shown in the "Balances" section, the NASA 1617 has a relatively high design load capability as compared to other 16-Foot balances (hence the term "high loads" system). As a result of the balance's large physical size, the outer sleeve of the bellows assemblies and the air supply ducts (2.10 dia.) through the balance block were required to be smaller than those found on the low loads system.

The low loads system (figure IV-11) was designed around the NASA 1621 (or NASA 1630) strain gage balance which has a smaller design load range than the NASA 1617 (see "Balances" section). The outer sleeve of the bellows and the air supply ducts (2.85 dia.) through the balance block are considerably larger (than those of the high loads system) allowing larger mass flows to be supplied to the nozzles. Except for the outer sleeve of the bellows assemblies, all bellows hardware are interchangeable between the two systems.

Both the high loads and low loads systems are housed inside the constant cross-sectional area fuselage of the wing-tip support system centerbody. This fuselage is a 5-by 10-inch rectangle with 1 inch radii on the corners. The metric break is located at FS 44.7. Afterbody hardware can be attached to the balance block at FS 49.3 as shown in figures IV-10 and 11. The wing-tip-support system forebody can be mounted on the NASA 835 force balance to obtain forebody forces and moments or can be grounded to the wing/centerbody when a metric forebody is not needed. As is the case with the afterbody, the forebody can easily be replaced with other designs.

The wing-tip-mounted twin jet propulsion system can also be installed on the sting-strut support system as shown in figure IV-12. High-pressure air is ducted into the high-pressure plenum which attaches to the top of the sting strut support system. Two air supply pipes, which are secured to each side of the plenum, duct the flow into the two separate flexible metal bellows assemblies as shown. Manually operated flow control valves in each supply pipe serve to balance the airflow to each bellows assembly. The bellows assemblies and balance block are the same hardware used on the high loads system. Because of space limitations, the main force balance used with this system is the NASA 1621 balance. As indicated in the sketch, the nonmetric portions of the external hardware are duplicates of the wing-tip-mounted support system fuselage (5-by 10-inch rectangular shape with 1.0 inch radii on corners) and forebody.

C. Flexible Bellows Systems.- The primary component of each of the 16-Foot Transonic Tunnel propulsion simulation systems is the flexible bellows system. At the present time, two different bellows design concepts are used: flexible metal bellows and nonmetallic (fabric) bellows. While both formed and welded metal bellows units have been tried, the welded fabrication technique has been the more successful metal bellows concept. Nonmetallic bellows are relatively new, however, early results with this type of bellows (ref. IV-6) are very encouraging. The diaphragms are fabricated, using an injection-mold technique, from a polyester based urethane. One advantage of the nonmetallic bellows is the low cost and fabrication lead time, relative to the metal bellows. The physical dimensions of these bellows vary with the particular concept and propulsion system used; however, the general arrangements are similar. A typical flexible metal bellows arrangement is presented in figure IV-13, and a sketch of the fabric bellows arrangement is shown in figure IV-14. In general, bellows are designed to eliminate any transfer of axial momentum as the air (for propulsion simulation) is passed from the nonmetric to metric portions of the model. As indicated in the figure, the high-pressure air is ducted through the supply pipe and is then discharged perpendicularly to the model axis through sonic nozzles equally spaced around the supply pipe. The two flexible bellows which connect the nonmetric to metric portions of the system are used as seals and serve to compensate for the axial forces caused by pressurization. The cavity between the supply pipe and bellows is vented to the model internal pressure.

The 16-foot bellows arrangements are limited to an operating pressure of 600 psi over a temperature range from 100°F to -50° F. The bellows are designed to deflect axially less than a maximum of .010 inches from the installed free length and have a maximum angular deflection of the model about the inner sleeve of less than 30 minutes.

D. Propulsion Simulation for Subsonic Transport Models.- Figure IV-15 presents photographs of the transport models with underwing, flow-through nacelles installed in the 16-Foot Transonic Tunnel used to study wing/pylon/nacelles integration for subsonic transports. The support sting and balance used for flow-through nacelle tests are respectively the drawing LD-515641A sting shown in figure II-32 and the NASA 838 strain gage balance. The same balance adapter and model nose hardware are used for both existing transport models for flow-through propulsion simulation (unpowered), and will also be used for a new low-wing higher sweep supercritical wing designed for M= 0.85. The high-pressure air supply sting used to support these models is shown in the sketch of figure IV-16 to indicate its method of attachment to the model and NASA 1627 balance (figures VI-11 and 12). Further information on the sting can be found in Section III on stings, and on drawing number LE-534599. Figure IV-16 also shows the general arrangement of related model equipment for the six-component NASA 1627 balance system utilizing high-pressure air for jet simulation or to drive turbofan simulators. By utilizing different adapter lengths and shapes at each end of the balance, models of many different shapes can be tested so long as they are of about the same wing area and same fuselage diameter. Further information on the 1/24-scale transport may be found in reference IV-7.

Another model used for studying powered propulsion integration is the large scale turboprop model. Figure IV-17 indicates the overall dimensions of the model including the placement of the Tech Development Model 1700B turboprop simulators. The model may be tested either with wing-mounted nacelles as shown in figure IV-17 or with aft-fuselage mounted nacelles (indicated by dashed lines on figure IV-17). However, the high-pressure air supply system for the model can only be used for one configuration or the other (not a combination of both configurations). The primary instrumentation for the model consists of (see figure IV-17): the main balance (1617 balance, drawing number LA-931392, or 1629 balance, drawing number LA-941975); counter-rotating balances (1633 CRF forward balance, drawing number LC-1033352, and 1633 CRA aft balance, drawing number LC-1033199); and pressure taps on the wing and pylon/nacelles. The attachment of the high-pressure air supply sting to the model and the main balance is shown in figure IV-18. Further information on the sting can be found in Section III on stings and on drawing number LTV 75-002760 (fig. III-17). Figure IV-18 also shows the major components and the general arrangement of the high-pressure air supply system. The actual high-pressure air supply system is far more complex than the sketch of figure IV-18 indicates, and further details and information on the air supply system can be found on drawing number LE-542206. Although this model was primarily designed for turboprop simulation, the model arrangement is flexible and it is possible to design nacelles for the model that utilize different types of propulsion simulation such as turbofan simulators or ducted fans.

E. <u>Turbofan Simulators</u>.- The high-pressure air driven turbofan simulator shown in the sketch of figure IV-19 was designed by Pratt and Whitney and built by Tech Development Inc. as model TD-460N. The engine has an air-driven, single-stage turbine that powers a single-stage 4.1-inch tip diameter compressor. At the design condition of 80,000 rpm, the turbine operates with a drive pressure of 350 psi and the fan pressure ratio is 1.6. Turbine drive gas requirements at this point are nominally 1.3 lb/sec at 350 psi and 160°F total pressure and total temperature, respectively. This pressure ratio and the ratio of fan diameter to turbine diameter are representative of current medium bypass ratio turbofan engines such as the P&WA JT9D.

The fan and turbine are supported on two lubricated ball bearings requiring oiling every 2 to 4 hours running time depending on orientation of the simulator. Each bearing has its own oil wick and annular oil tank which is filled using a hypodermic syringe to inject approximately 6 cc of MIL-L-7808c high quality turbine oil through fittings in the pylons. Bearing temperatures are measured by two iron/constantan thermocouples per bearing. Steady state operating temperatures at 80°F and 80,000 rpm are 160°F for the front bearing and 60°F for the rear bearing. Front bearing temperature will increase with tunnel total temperature while rear bearing temperature can be controlled using drive air temperature control. Bearings are capable of operation up to 300°F but should not approach this temperature under normal conditions. Two magnetic pickups sensing six pulses/revolution from six magnetic pins imbedded in the aft end of the rotor are used to measure rpm of the simulators.

Important dimensions, mounting requirements for the simulator (through the pylon), and mounting diameters for the inlet, aft fan and turbine cowls are also given in figure IV-19 as a guide to users. Other turbofan simulators with similar operating characteristics, but of different scale or bypass ratio, may become available as needed.

Figure IV-20 presents a typical fan map for the TD-460N simulators obtained in a heavily instrumented Static Test Facility located near the 16-Foot Transonic Tunnel. Static pressures for the various Mach numbers were achieved using a high-pressure-air powered ejector to evacuate a chamber into which the fan and turbine flows exhausted. The airflow entering the inlet of the simulator is always deficient relative to the full-scale engine, because the air that drives the turbine and leaves the turbine nozzle enters from an external source. So, if the fan-nozzle area is simulated the airflow entering the inlet for any fan pressure ratio will be deficient by 1/(1 + BPR), where BPR is the bypass ratio. Existing inlet hardware for the 1/24-scale transport model (see previous section) have been sized for on-design operation at M = 0.8, 80,000 rpm and flowing only the fan mass-flow.

F. Turboprop Simulators.- For turboprop simulation, there are two types of air-driven motors available for testing, a single rotation motor (see figure IV-21) and a coaxial counter-rotating motor (see figure IV-23). Both types of motors were designed and built by Tech Development Inc., and designated as Model 666H for the single rotation motor and Model 1700B for the counterrotating motor.

The Model 666H is a high power-to-weight ratio, four-stage turbine. Power is derived from compressed air introduced into the first stage nozzle plenum via thirty radial supply ports. Four turbine stages are used to provide the power at the proper speed without the use of gears. The design operating point is an output of 195 horsepower at 16,000 rpm. The corresponding drive flow conditions are: an inlet total pressure of 500 psig;

an inlet total temperature of 60°F; and an inlet air mass flow of approximately 3.5 lbm/sec. Operational limits of the motor are defined as: a maximum speed of 16,000 rpm; a maximum drive air total pressure of 500 psig; and a maximum drive air total temperature of 150°F.

The rotating elements of the Model 666H motor are supported by two lubricated ball bearings requiring constant lubrication while motor is being used. The lubricating oil (MIL-L-7808) should be filtered before use to prevent blockage of the 0.014 inch diameter oil injection port at each bearing location. The oil is delivered to the bearings by a positive displacement pulse oil system. Under normal operating conditions, the oil delivery is approximately 10 cc/hr per bearing.

Bearing temperatures on the Model 666H motor are monitored by a single iron/constantan thermocouple per bearing. The front bearing will be affected by tunnel total temperature while the rear bearing is affected by the drive air temperature. The bearing temperatures should not exceed 180°F on the front bearings and 120°F on the rear bearings. A single magnetic pickup sensor monitors the shaft speed with thirty pulses per revolution generated by teeth on the air-motor shaft.

Figure IV-21 shows the Model 666H motor envelope dimensions and the major interior elements. The motor is mounted by a series of axial tapped holes in the forward face. An annular air supply plenum is required for installing the motor to communicate to the radial air supply ports in the motor housing. The outside diameter of 3.72 inches forms the inner wall of this annular plenum supplying the turbine drive air for the radial inlet to the first stage turbine plenum.

Figure IV-22 shows the Model 666H performance curves for various drive air conditions with an ambient pressure discharge of 14.7 psia.

The Model 1700B motor is a four-stage counterrotating air turbine with coaxial output shafts. The forward two turbine stages drive the outer shaft while the aft two turbine stages are coupled to the inner shaft. Both shafts rotate around a hollow central stationary shaft which is provided to route instrumentation from a fairing adjacent to the inner shaft fan hub. The drive air is injected through six radial ports and interstage ports are provided for trim air to balance the power output at off-design operating conditions. The design operating point is an output of 110 horsepower per shaft at 15,400 rpm. The corresponding drive flow conditions are: an inlet total temperature of 700 psia; an inlet total temperature of 100°F; an inlet air mass flow of 6.5 lbm/sec; and an exhaust static pressure of 228 psia. Operational limits of the motor are defined as: a maximum speed of 18,500 rpm per shaft; a maximum drive air total pressure of 800 psia; a maximum drive air total temperature of 150°F; a minimum exhaust total temperature of -40°F; and a maximum shaft power differential of 30 horsepower.

The stationary shaft of the Model 1700B motor is supported by two ball bearings which are permanently lubricated with Barden G-18 grease. The inner and outer shafts are supported by two ball bearings per shaft which are lubricated by a pulse oil jet system external to the motor, using MIL-L-7808 oil. Under normal operating conditions, the oil delivery is approximately 10 cc/hr per bearing.

Bearing temperatures on the Model 1700B motor are measured by two chromel/alumel thermocouple per bearing. The rotating shaft bearing thermocouples are mounted to read outer race temperatures while the stationary shaft bearing thermocouples read inner race temperatures. The outer race is also instrumented with two resistance temperature detectors (RTD) which rotate with the outer shaft. The bearing temperatures should not exceed 200°F on the rotating shaft bearings and 270°F on the stationary shaft bearings. Two magnetic pickup sensors per shaft monitor shaft speed with 30 pulses per revolution generated by teeth on the turbine shafts.

Figure IV-23 shows the Model 1700B motor envelope dimensions and the major interior elements. The motor is mounted by a series of axial tapped holes in the forward face. Annular air supply plenums are required in the installation to provide ducts to the radial ports of the main turbine air inlets and the trim air inlets. O-ring grooves are provided on the motor housing surface to seal these flow paths.

Figure IV-24 shows the Model 1700B performance curves at various drive air conditions.

References

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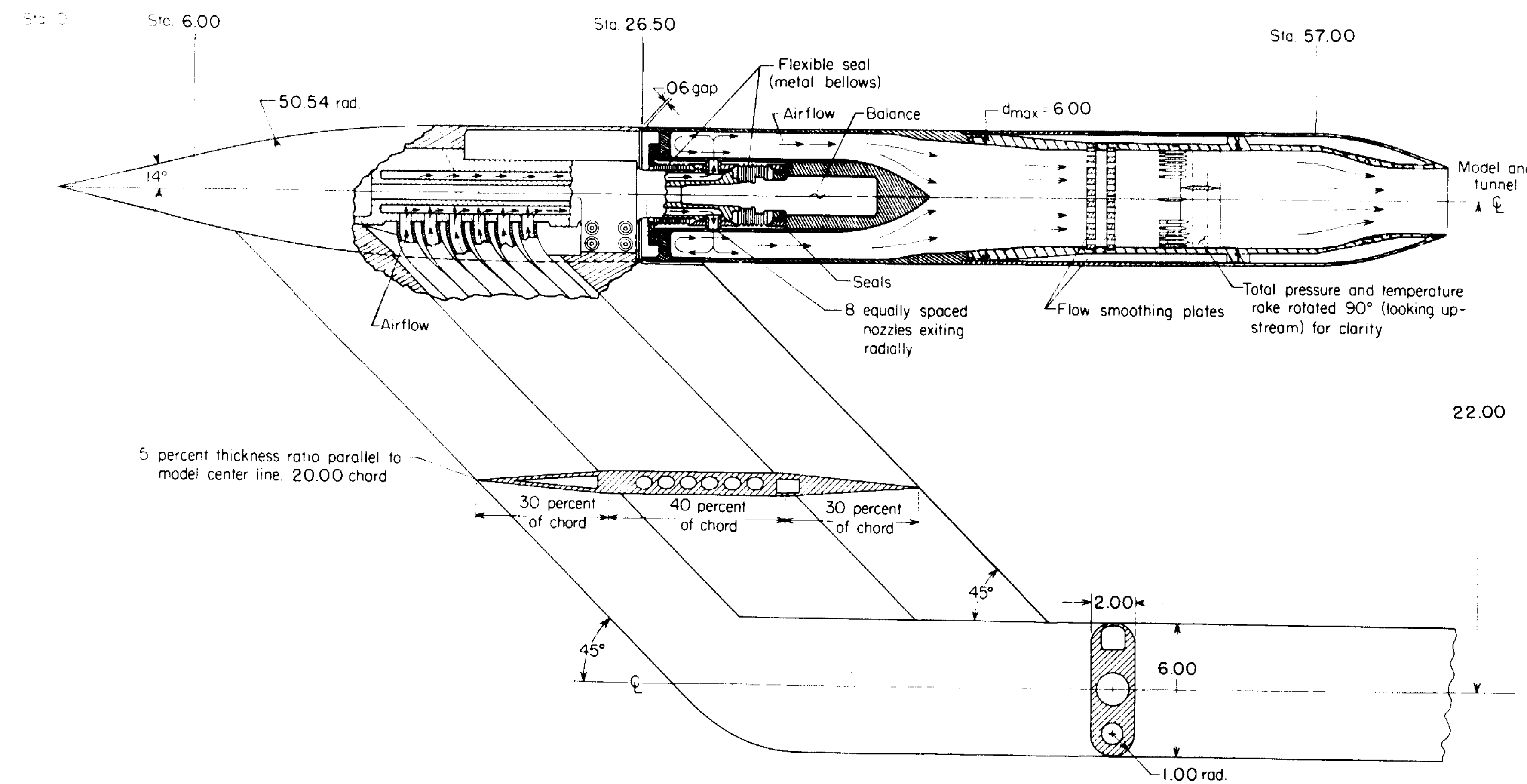

Figure IV-1. Sketch of axisymmetric single-engine propulsion simulator with an axisymmetric convergent-divergent nozzle installed. (All dimensions are in inches.)

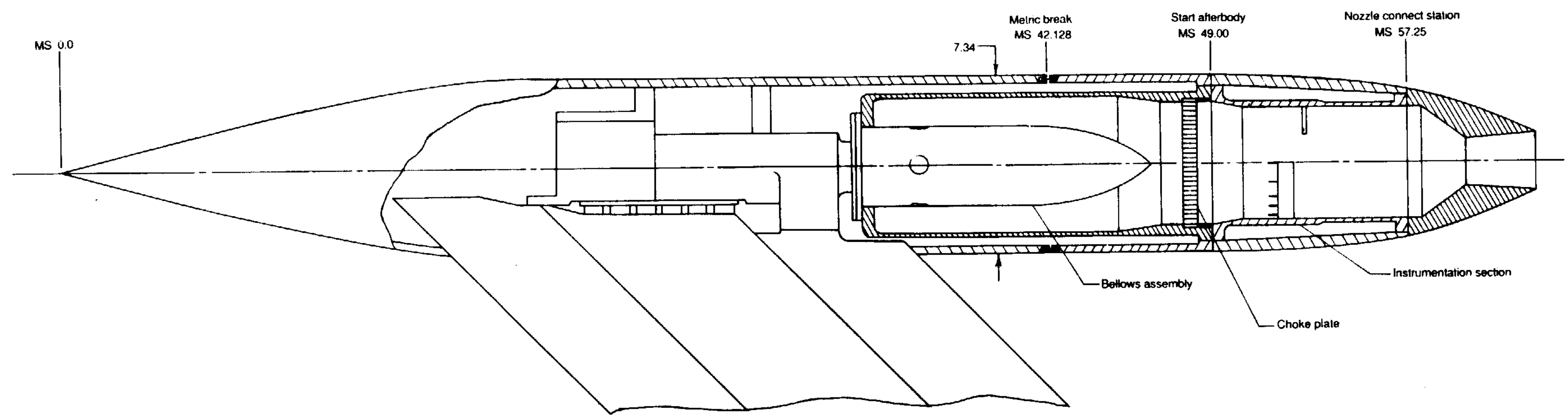

Figure IV-2. Sketch of large diameter axisymmetric single-engine propulsion simulator. All dimensions are in inches.

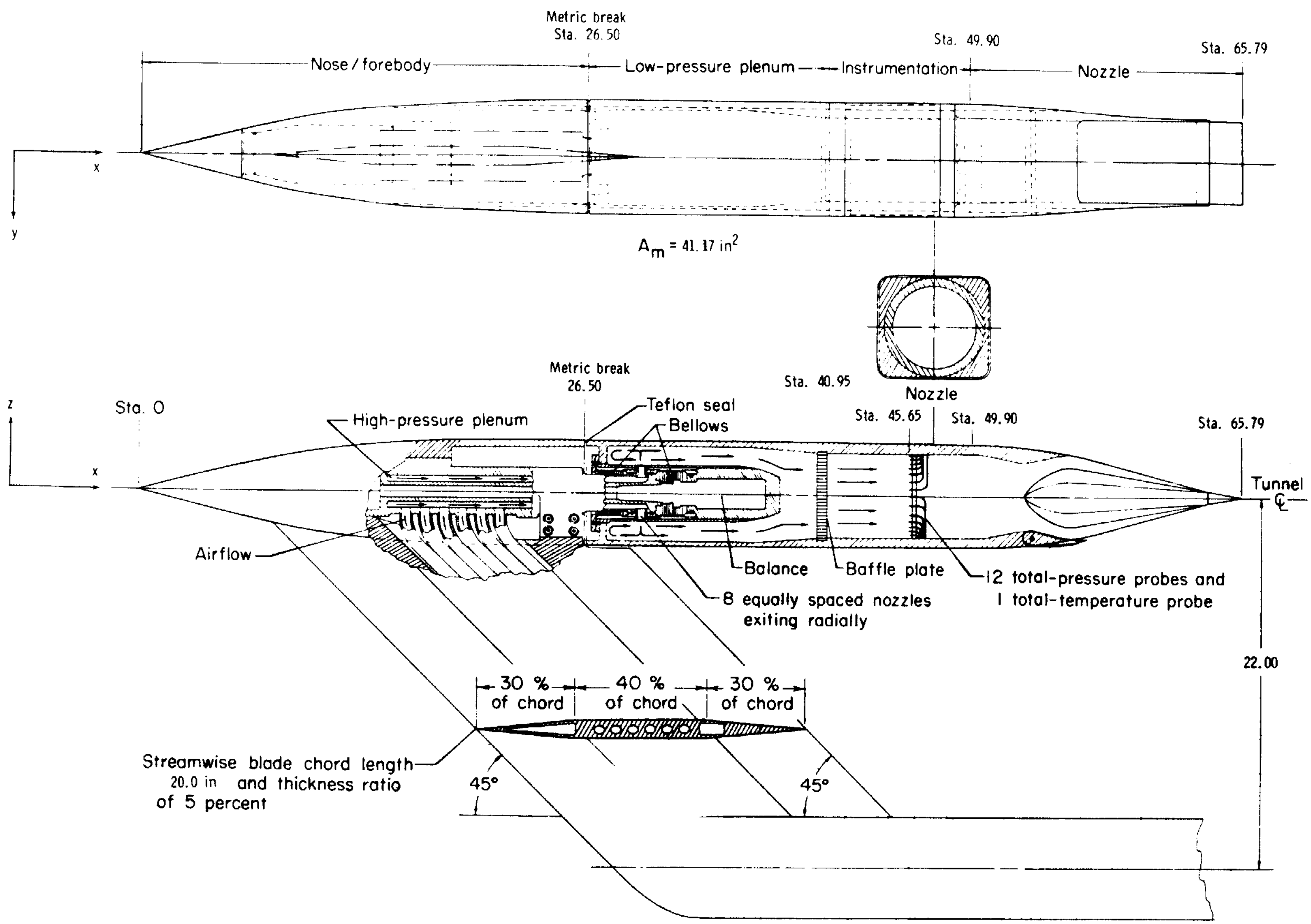

Figure IV-3. Sketch of nonaxisymmetric single-engine propulsion simulator with two-dimensional wedge nozzle installed. (All dimensions are in inches.)

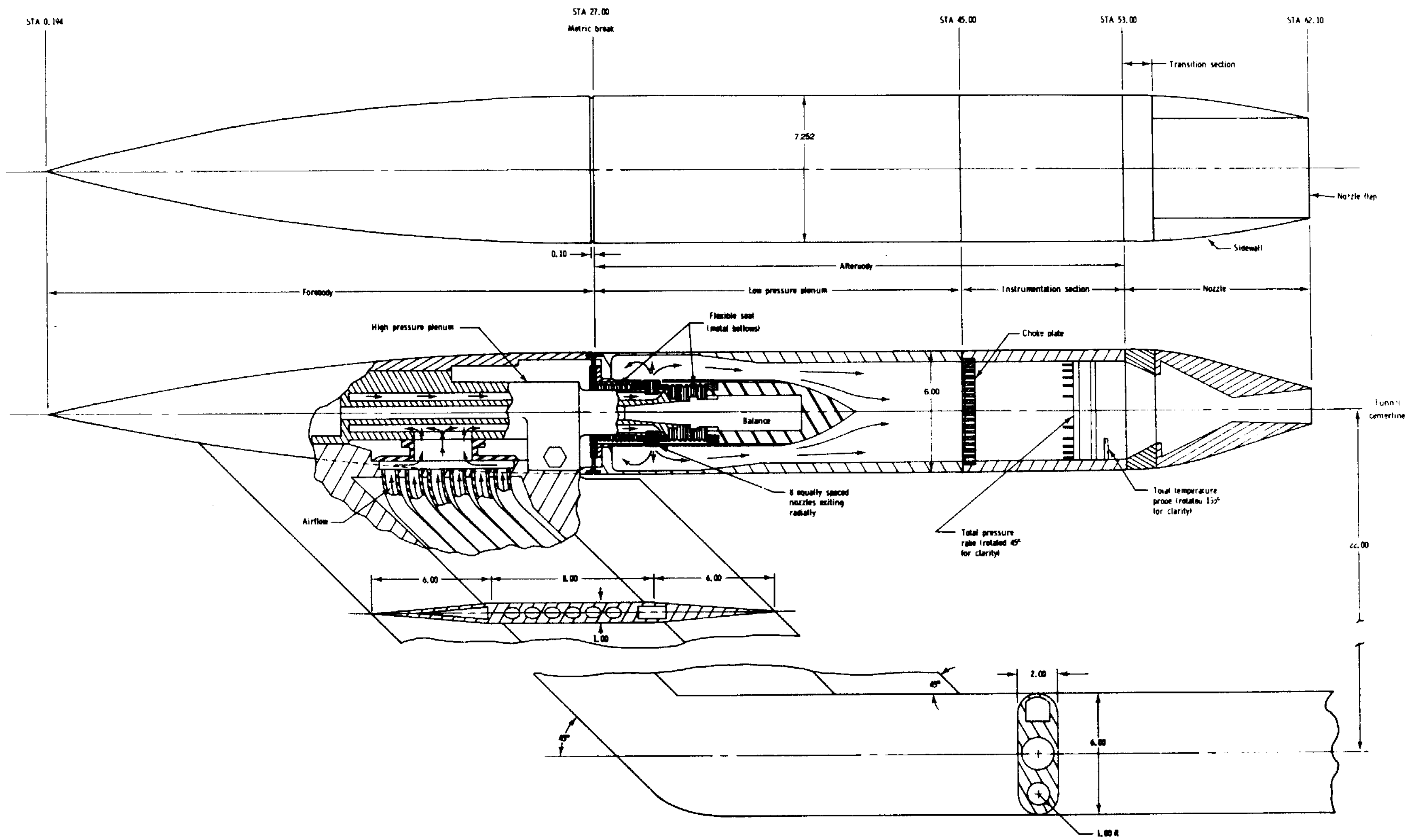

Figure IV-4. Sketch of single-engine propulsion simulator with nonaxisymmetric body number 1 installed.

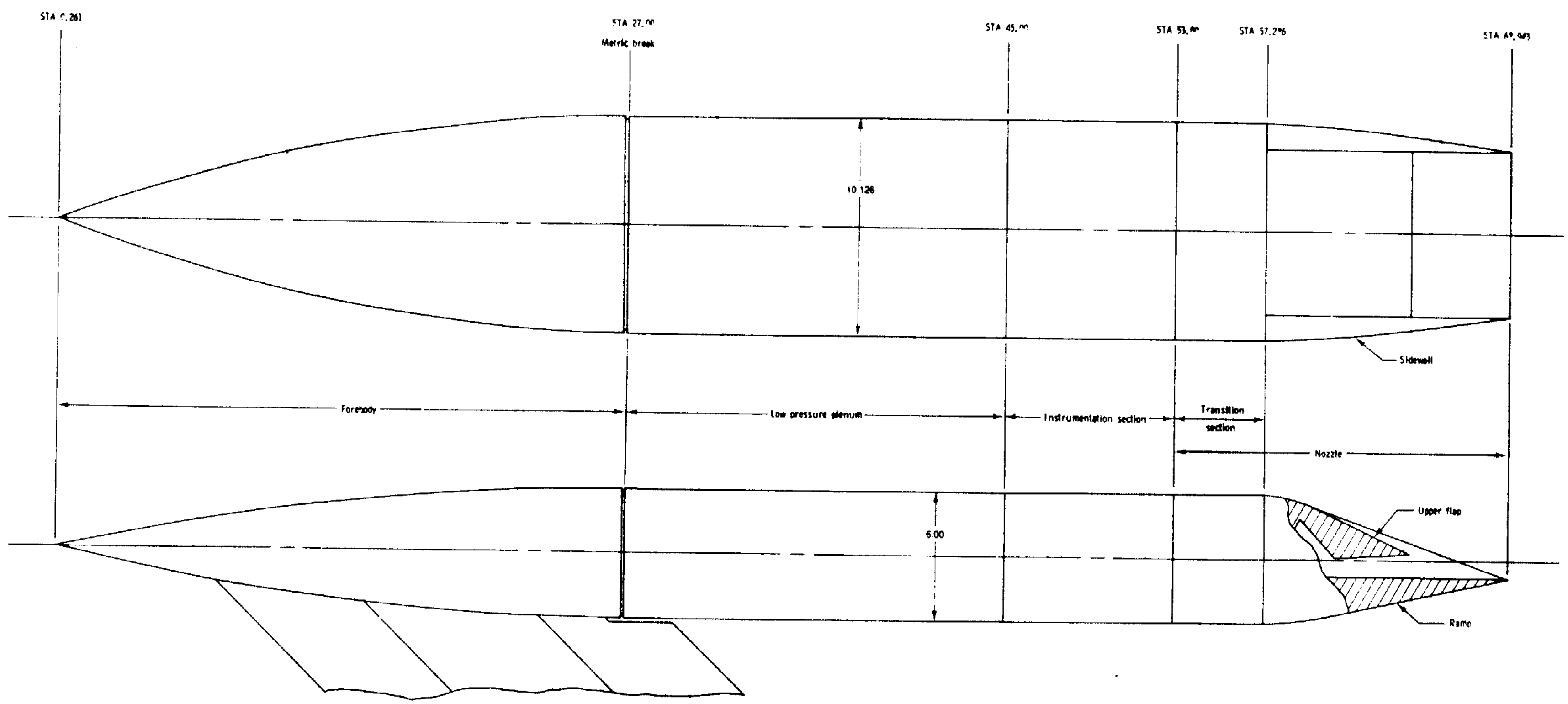

Figure IV-5. Sketch of nonaxisymmetric body number 2.

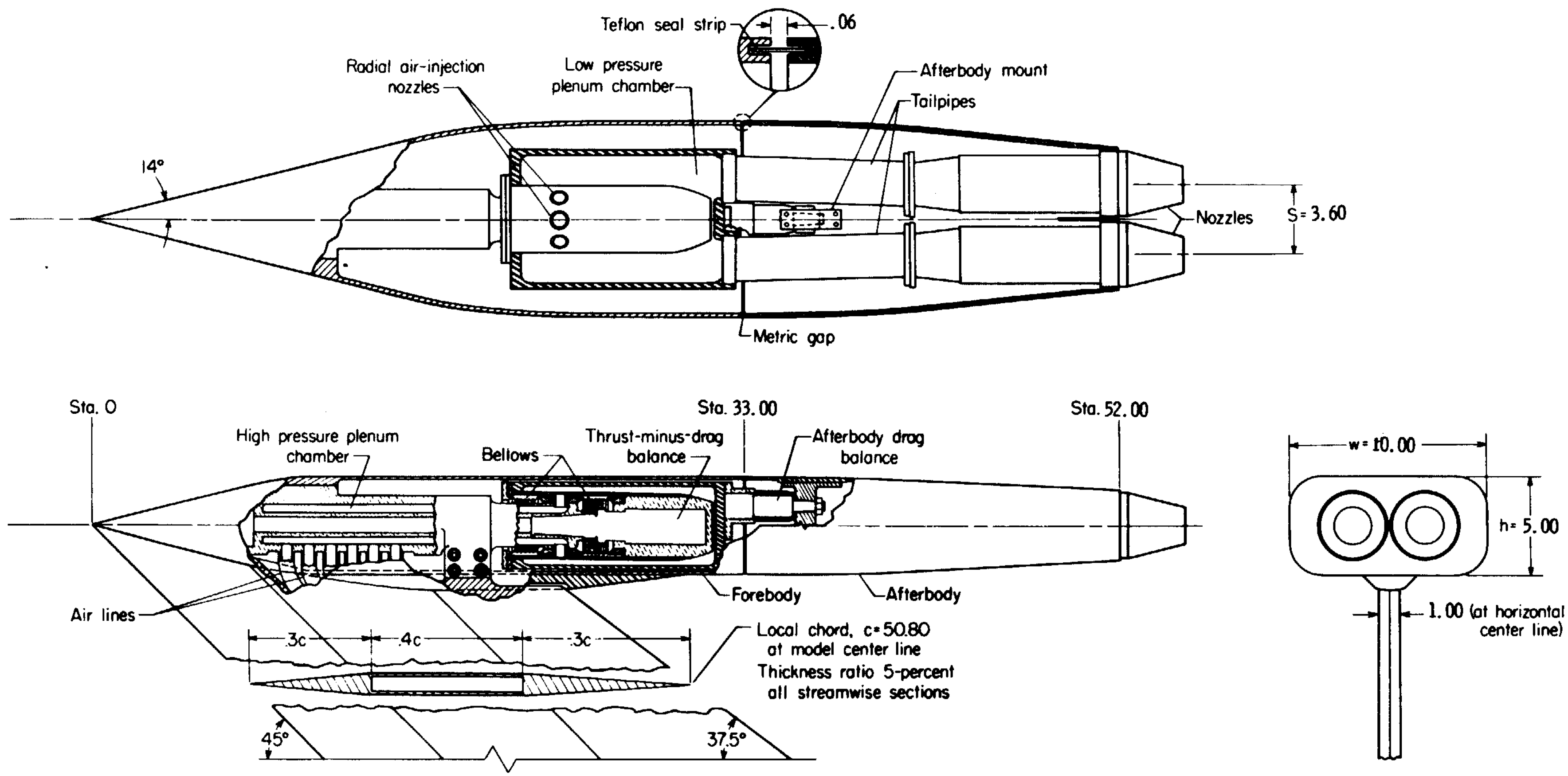

Figure IV-6. Sketch of single bellows twin-engine propulsion simulator with a typical axisymmetric afterbody installation. (All dimensions are in inches.)

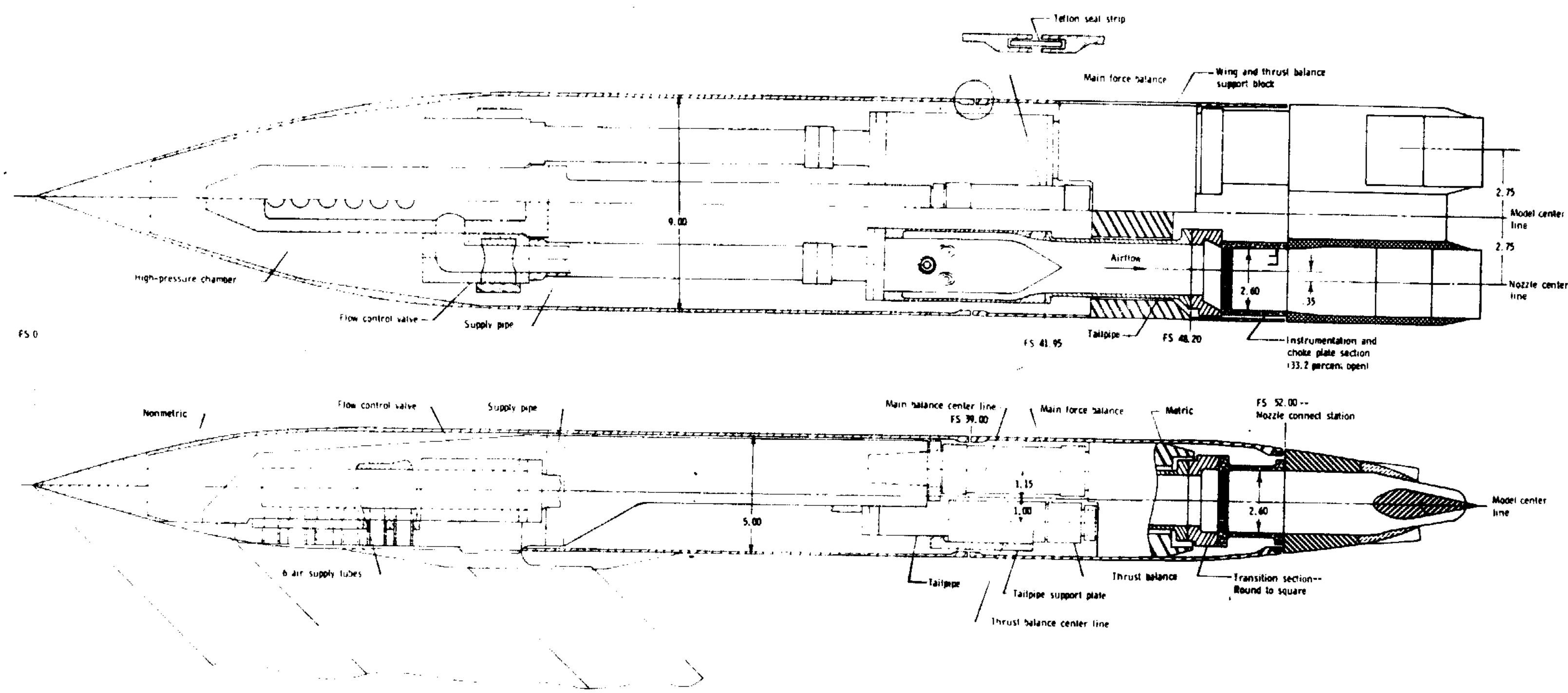

Figure IV-7. Sketch of the twin-jet propulsion simulation system with a typical nonaxisymmetric nozzle installation. (All dimensions are in inches.)

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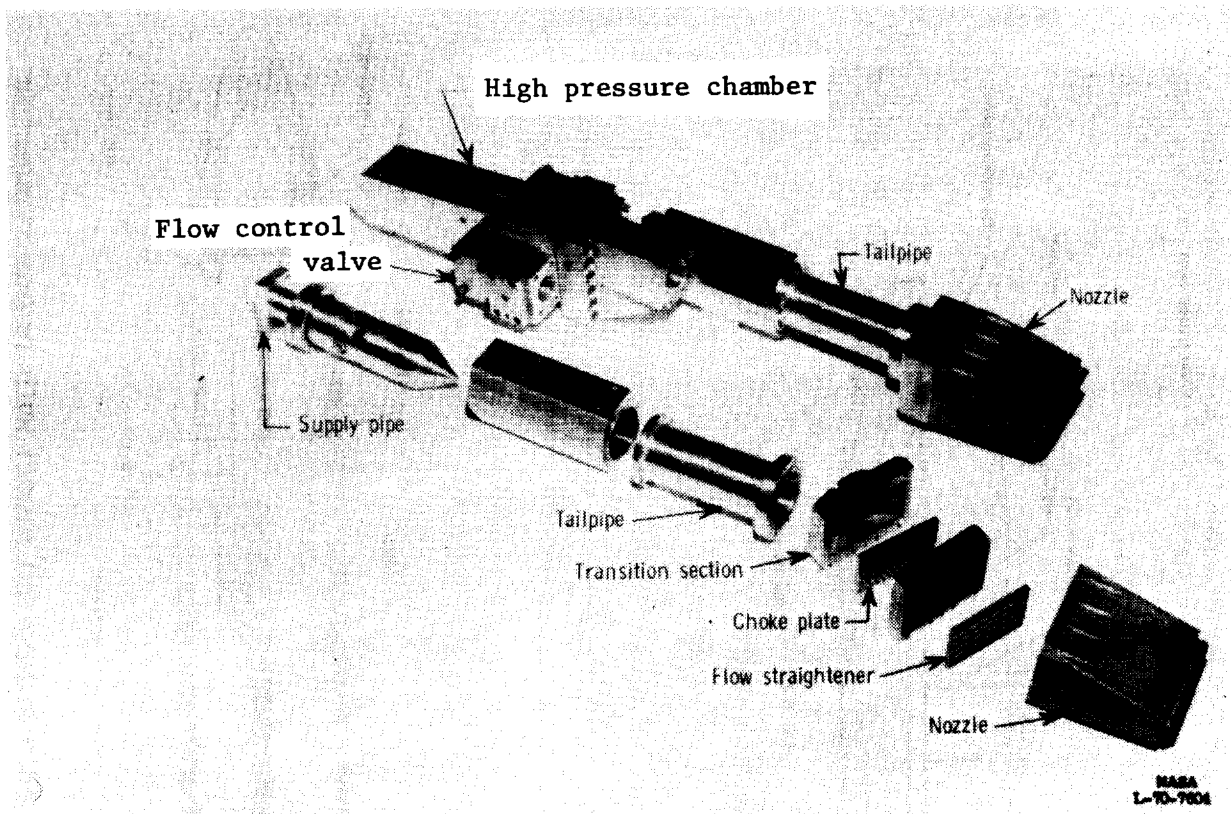

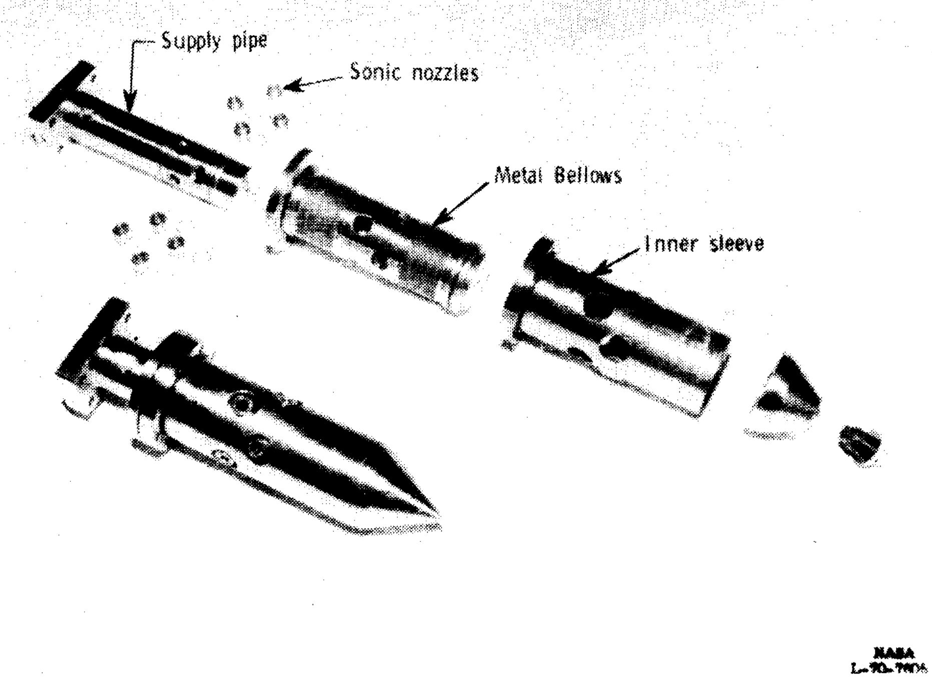

Figure IV-8. Photographs of twin jet simulator internal hardware.

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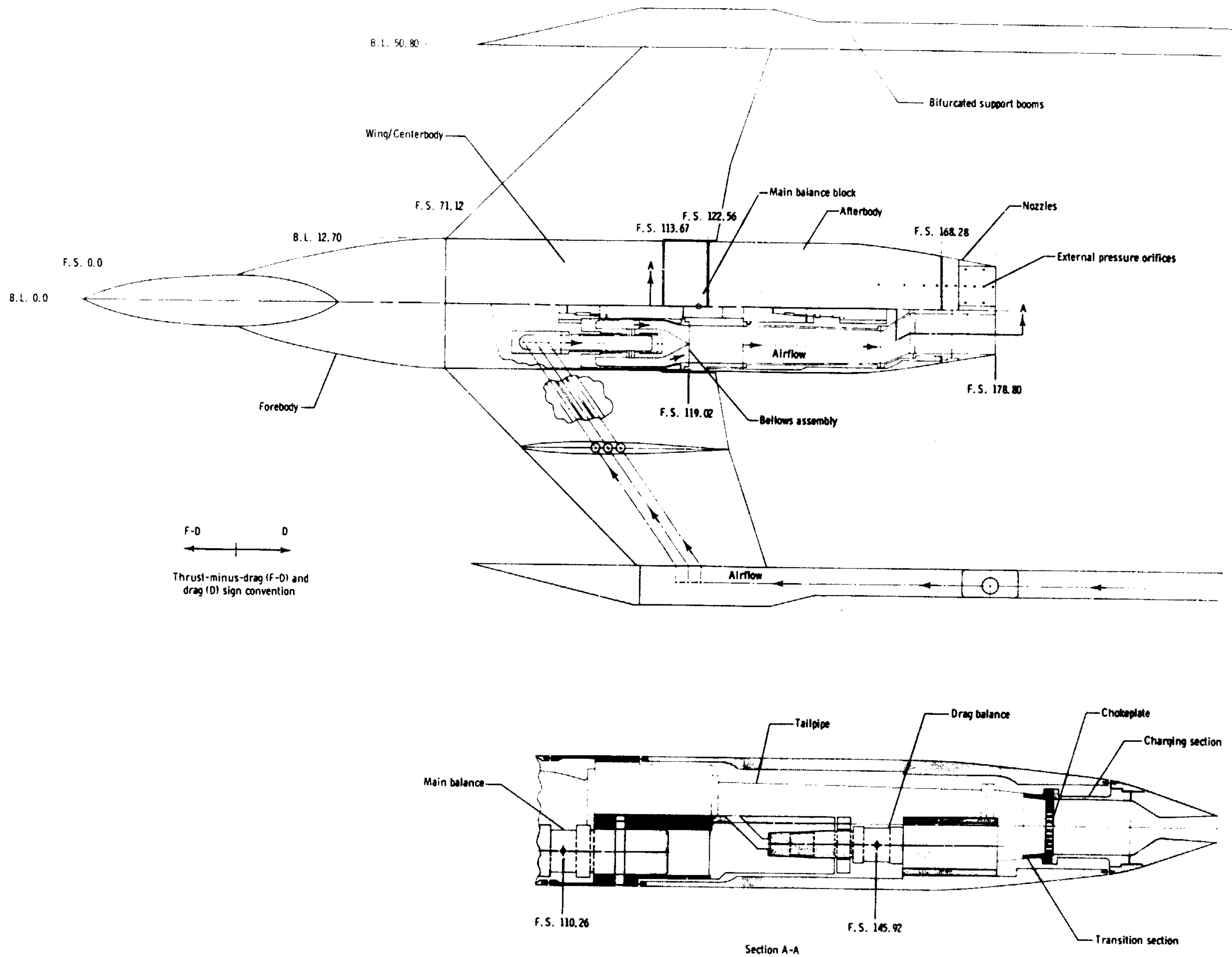

Figure IV-9. Sketch of air powered, twin engine, wing-tip supported model with nonaxisymmetric convergent-divergent dry power nozzles showing jet simulation system and balance arrangement.

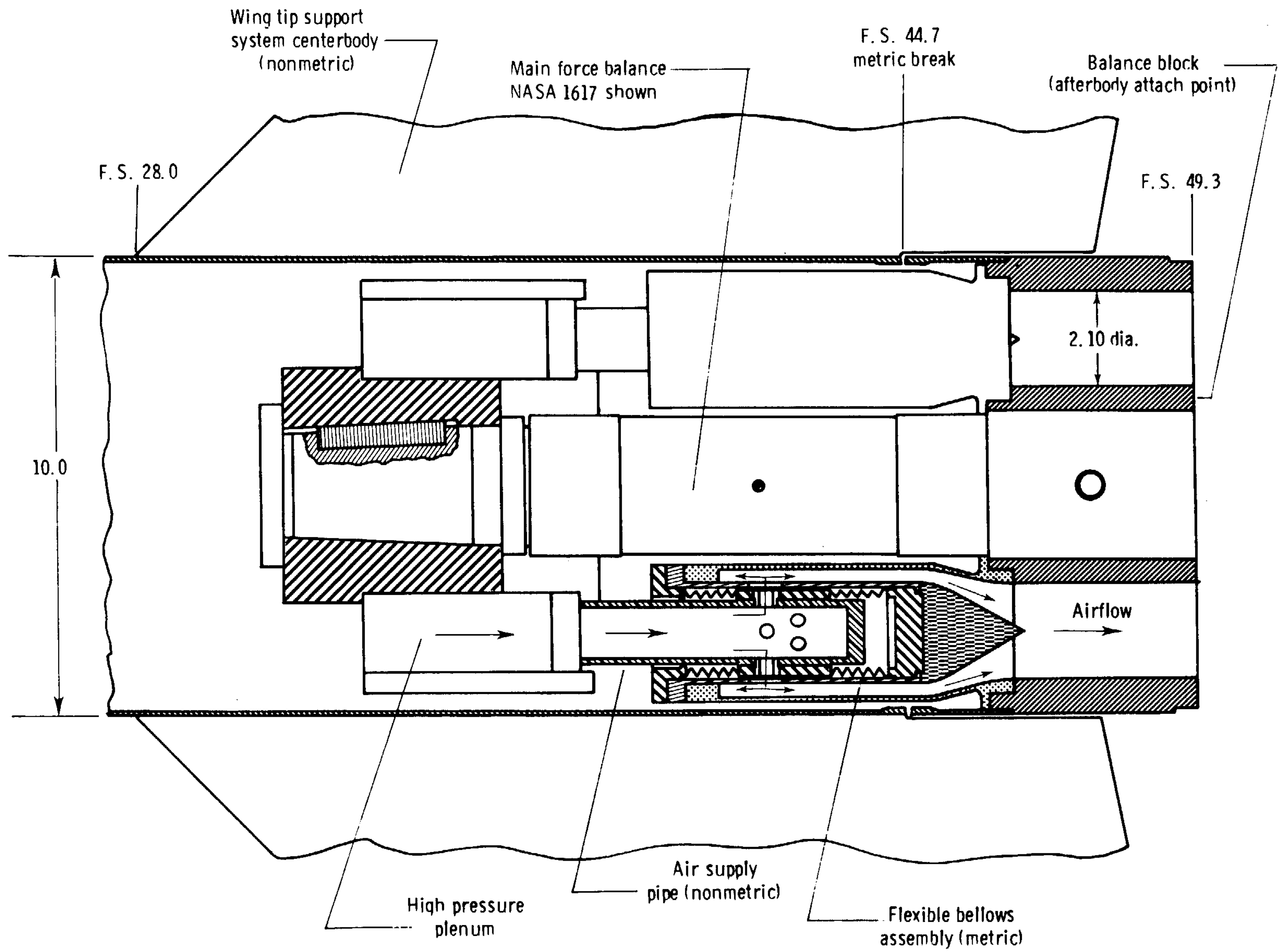

Figure IV-10. Sketch of wing-tip-mounted twin-jet simulator showing the high loads system. (All dimensions are in inches.)

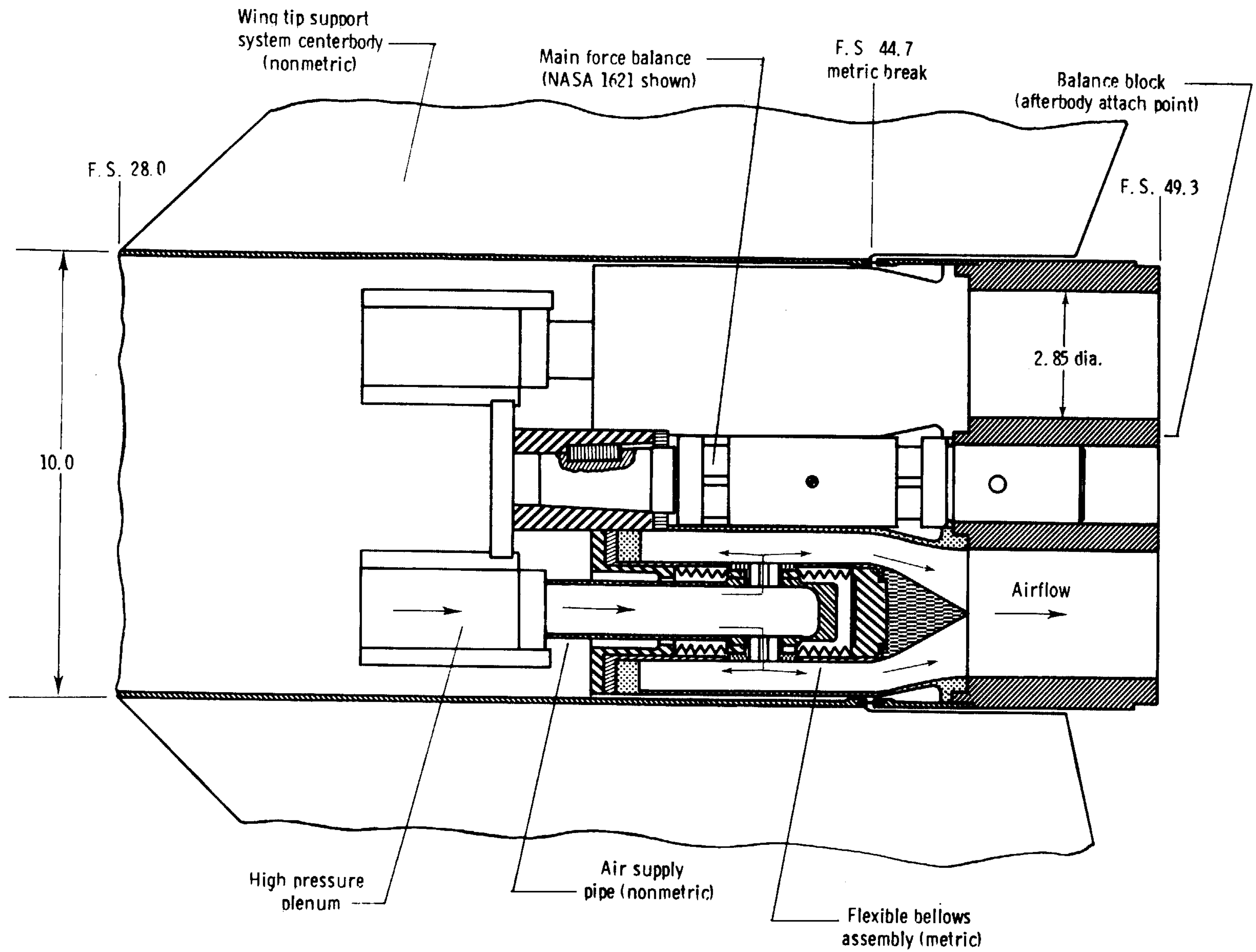

Figure IV-11. Sketch of wing-tip-mounted twin-jet simulator showing low loads system. (All dimensions are in inches.)

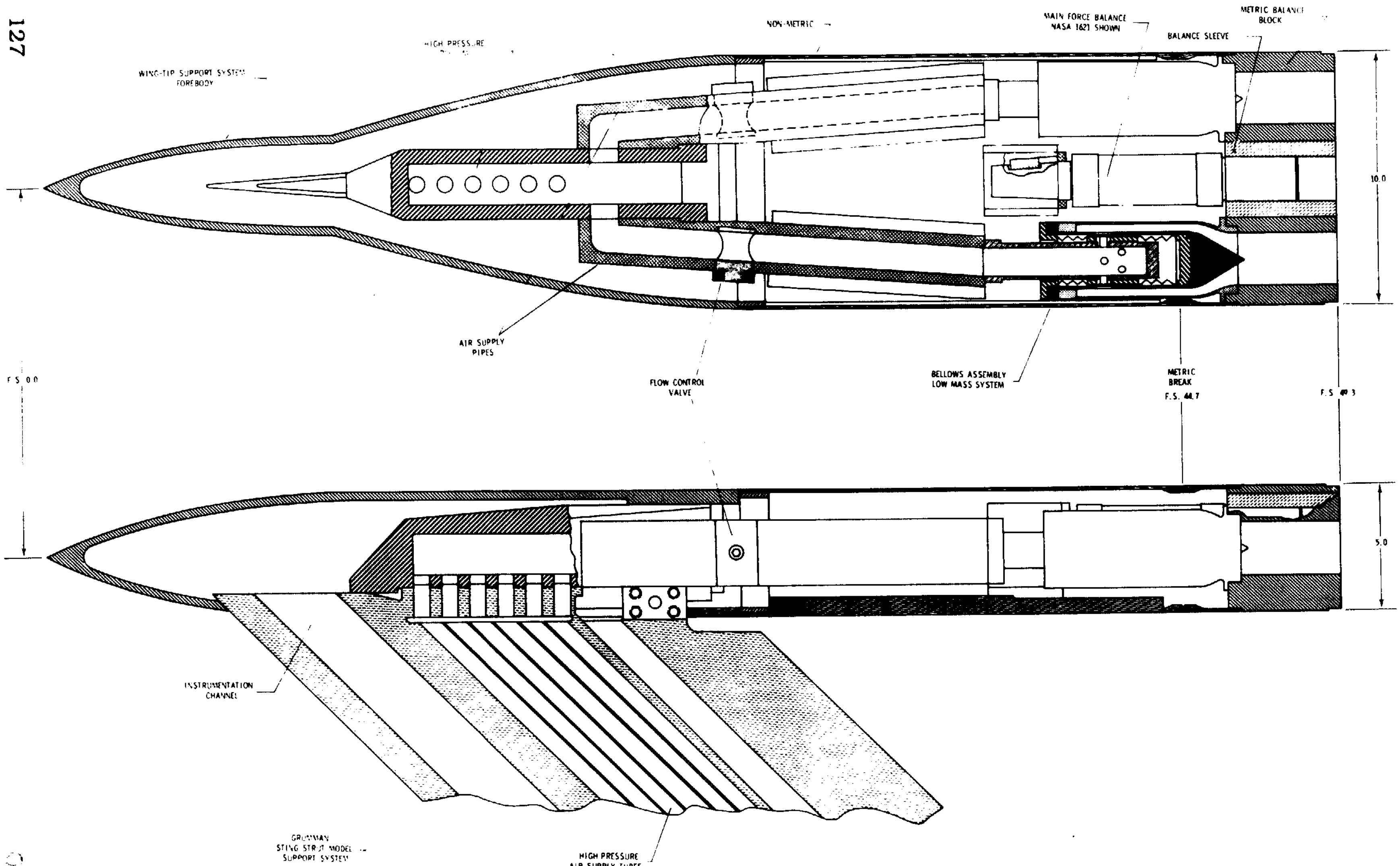

Figure IV-12. Wing-tip-mounted twin-jet simulation system installed on sting-strut support system. (All dimensions are in inches.)

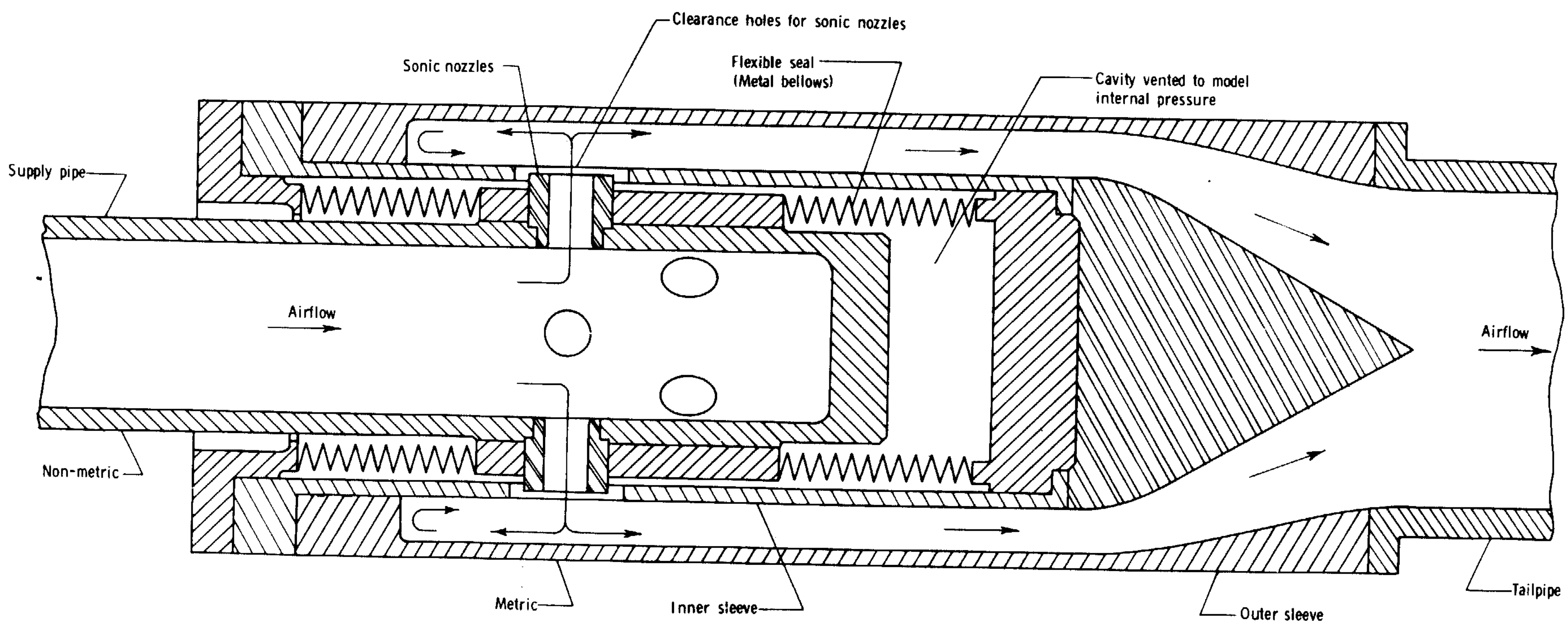

Figure IV-13. Sketch of typical flow transfer assembly showing the flexible metal bellows.

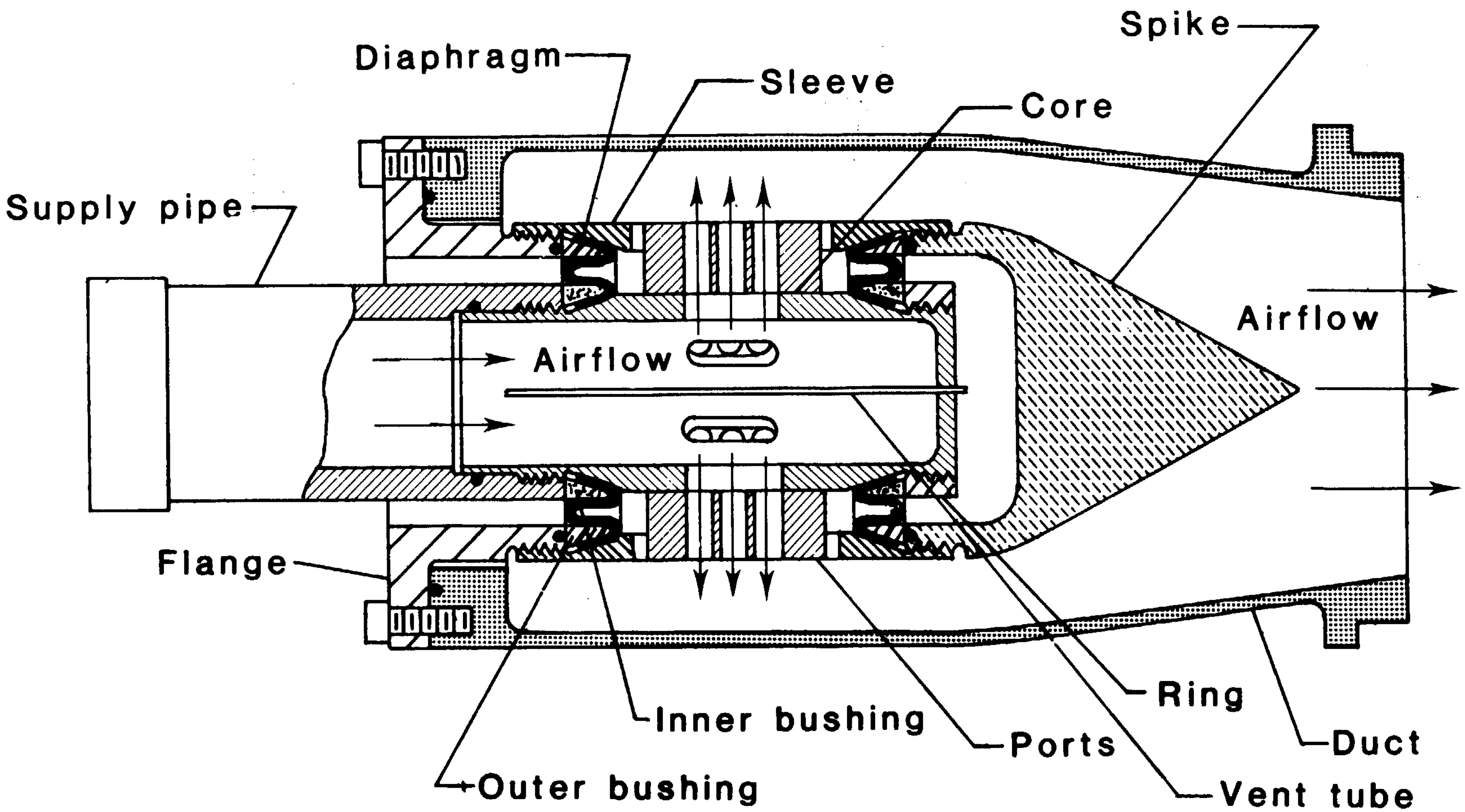

Figure IV-14. Sketch of typical flow transfer assembly showing fabric bellows installed.

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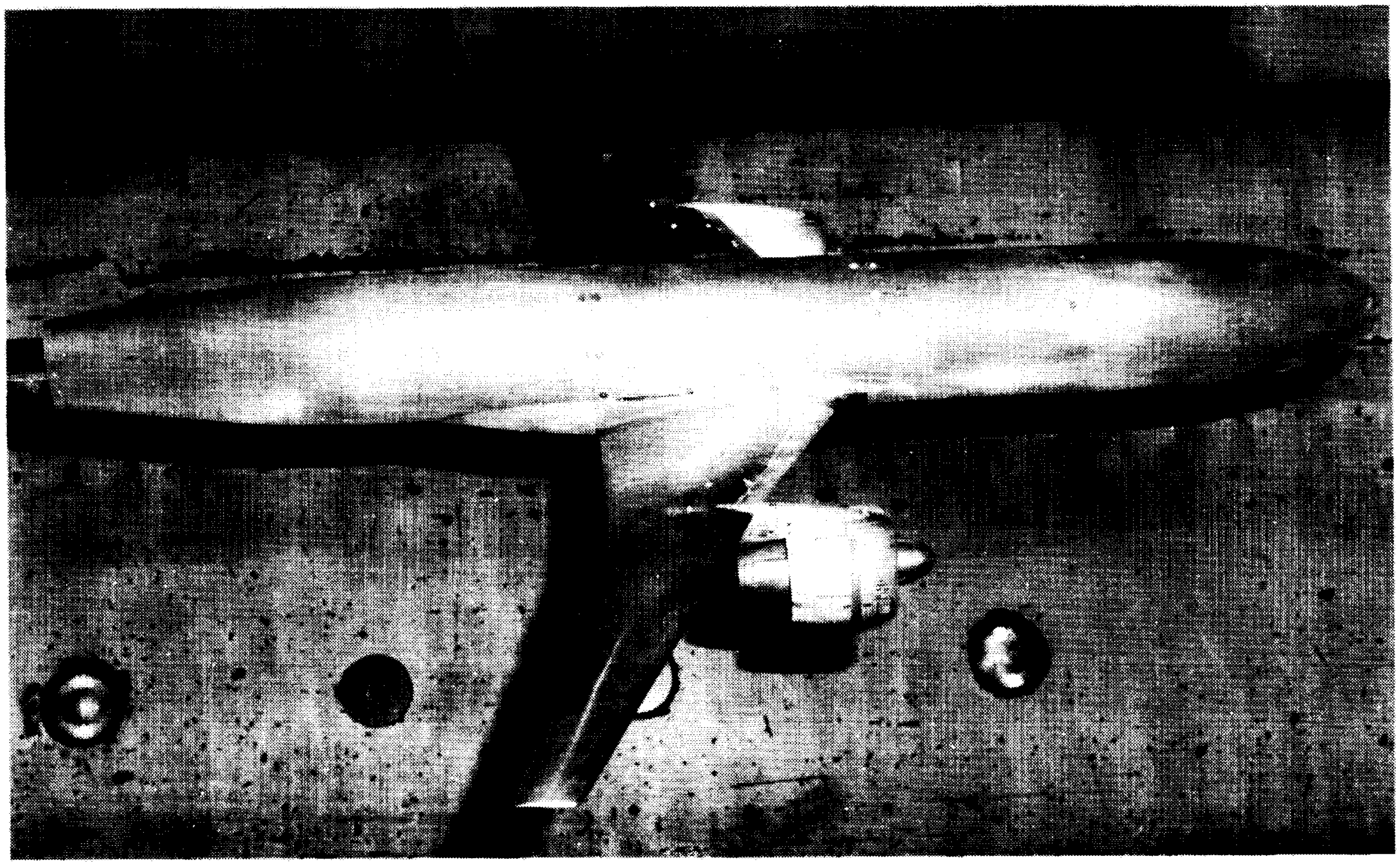
(a) 1/24-scale high wing transport model with flow through underwing symmetrical nacelles.

Figure IV-15. Photographs of the transport models.



(b) 1/17-scale low-wing transport model with flow through superfan nacelles.

Figure IV-15. Concluded.

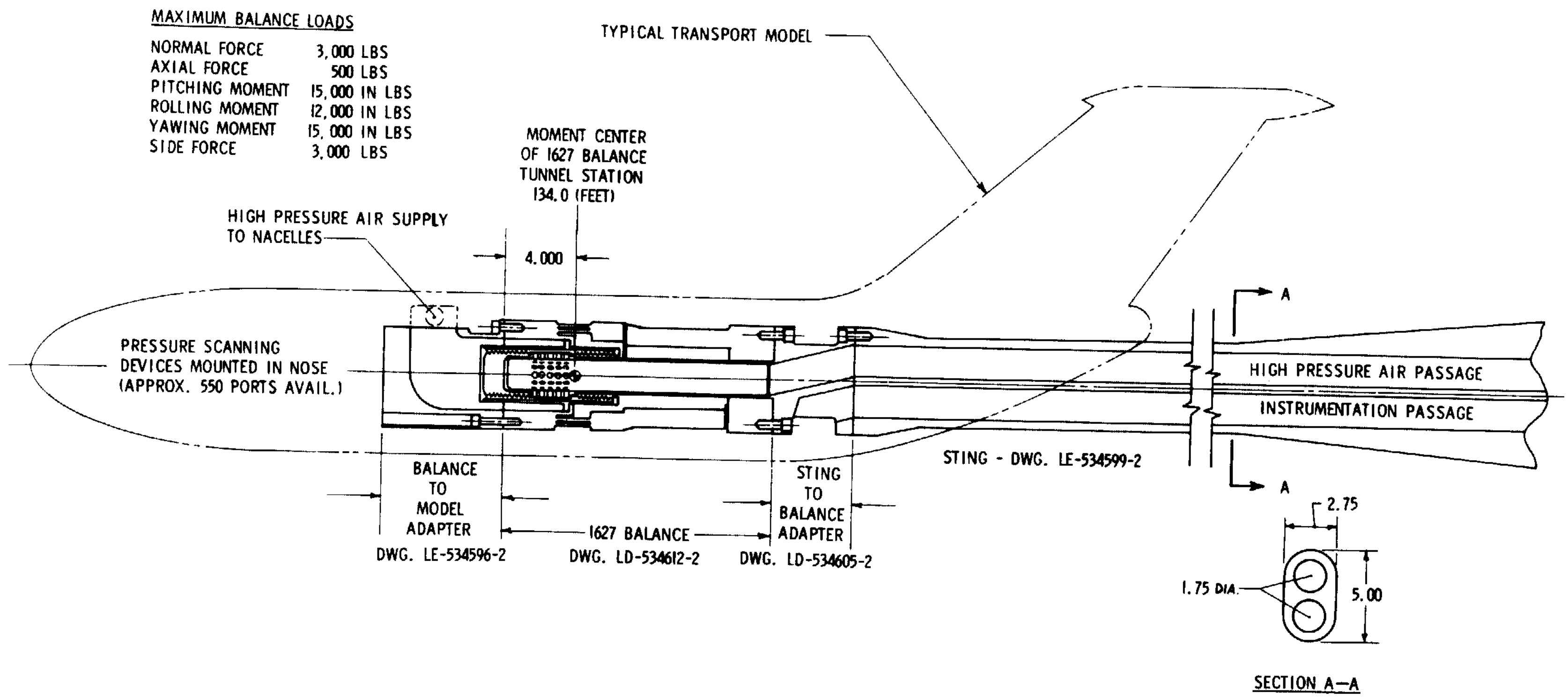


Figure IV-16. Sting-balance system for turbofan simulators. (All dimensions are in inches.)

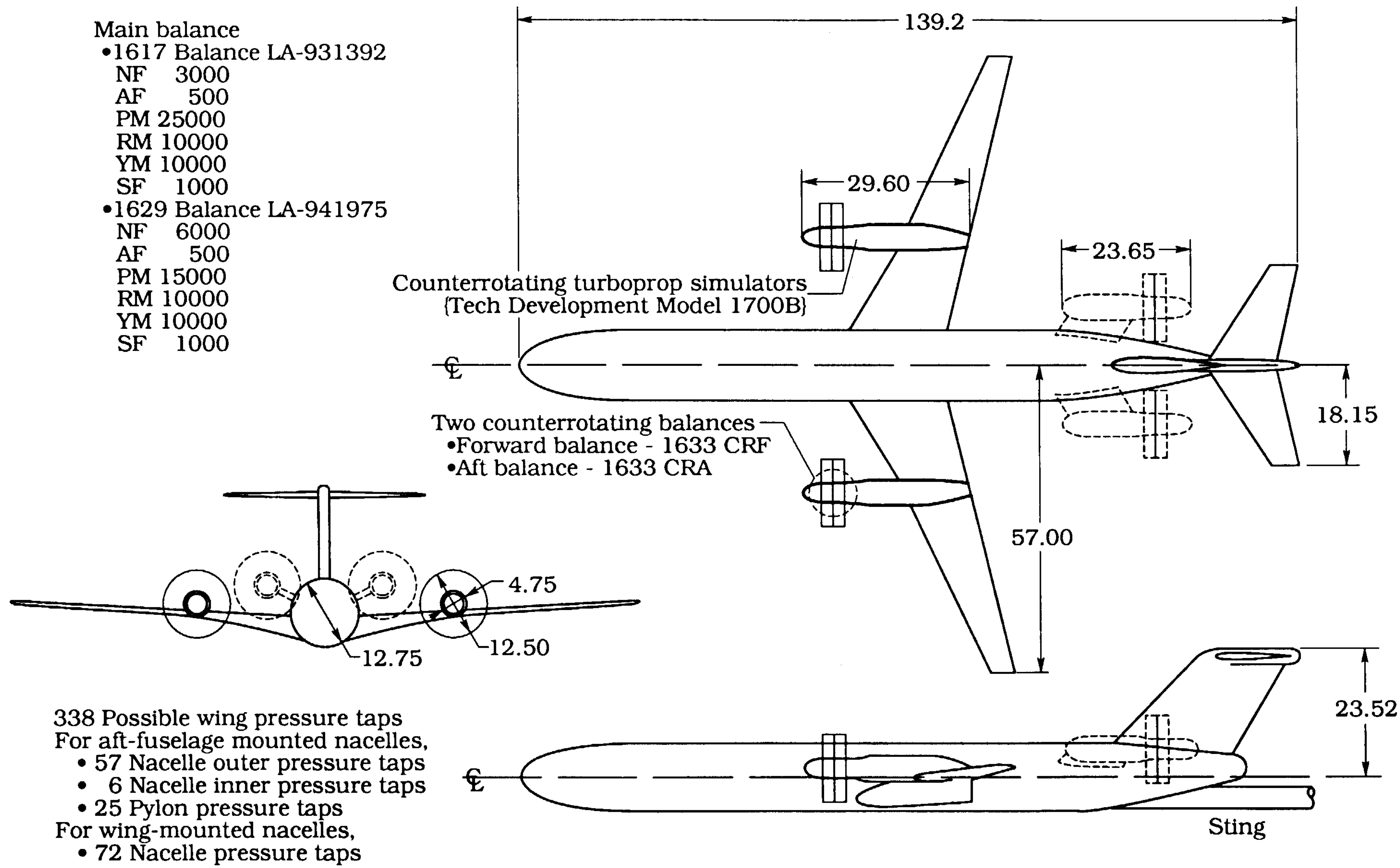


Figure IV-17. Transport Model with Turboprop Simulators (LE-542201). (All dimensions are in inches.)

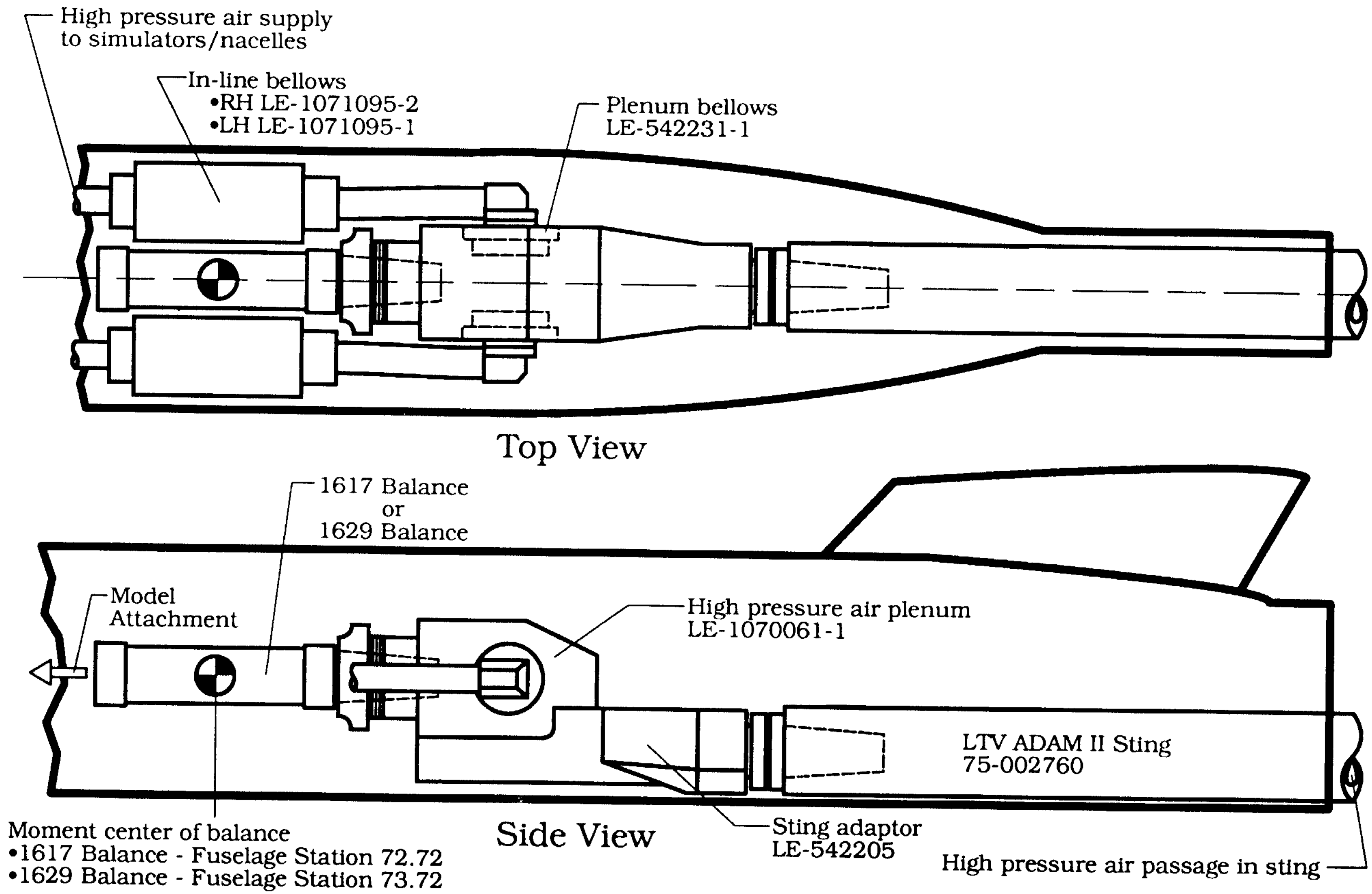


Figure IV-18. - Sting - Balance System and Air Supply System for Turboprop Model.
All dimensions are in inches.

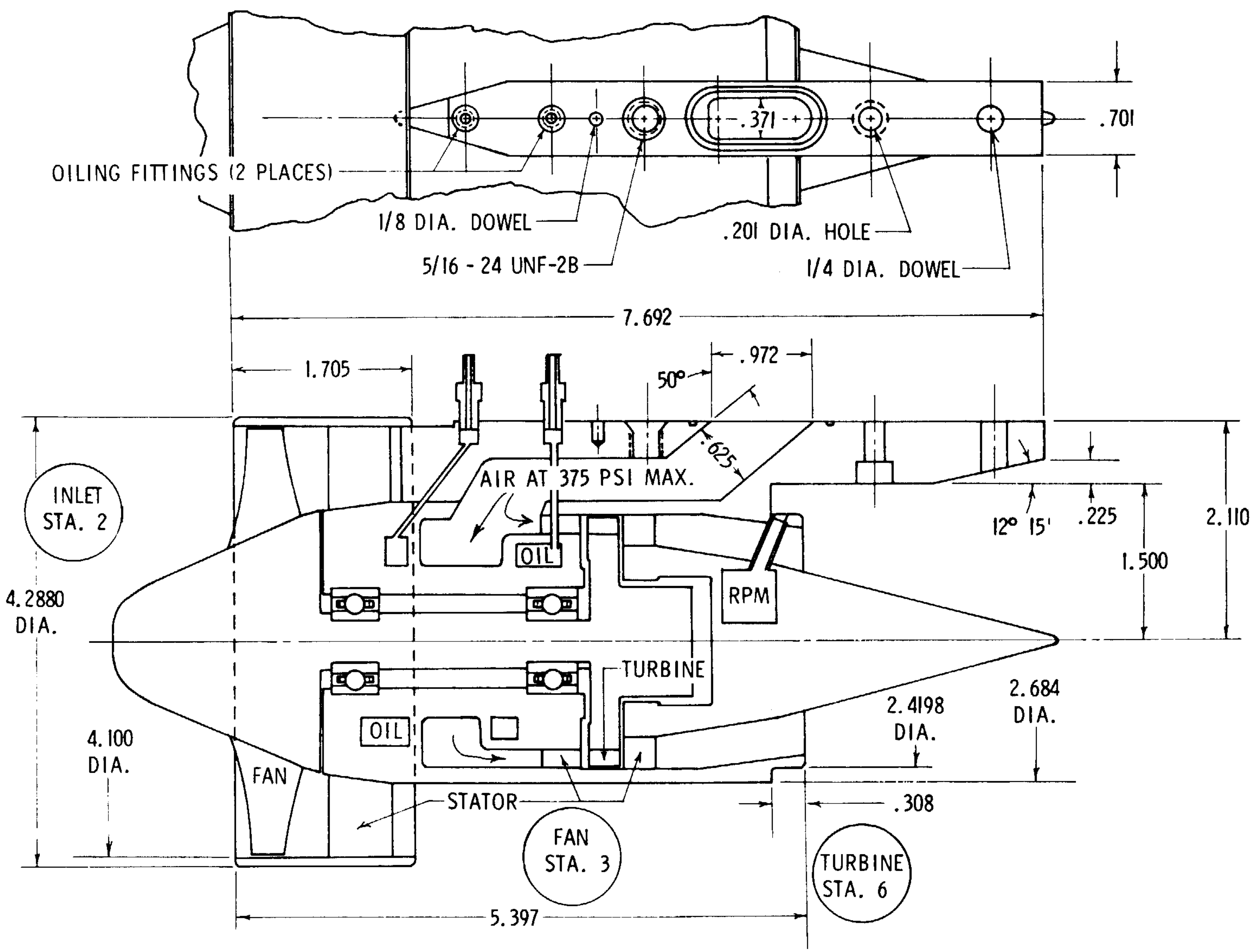


Figure IV-19. 4.1-inch turbofan simulator. (All dimensions are in inches.)

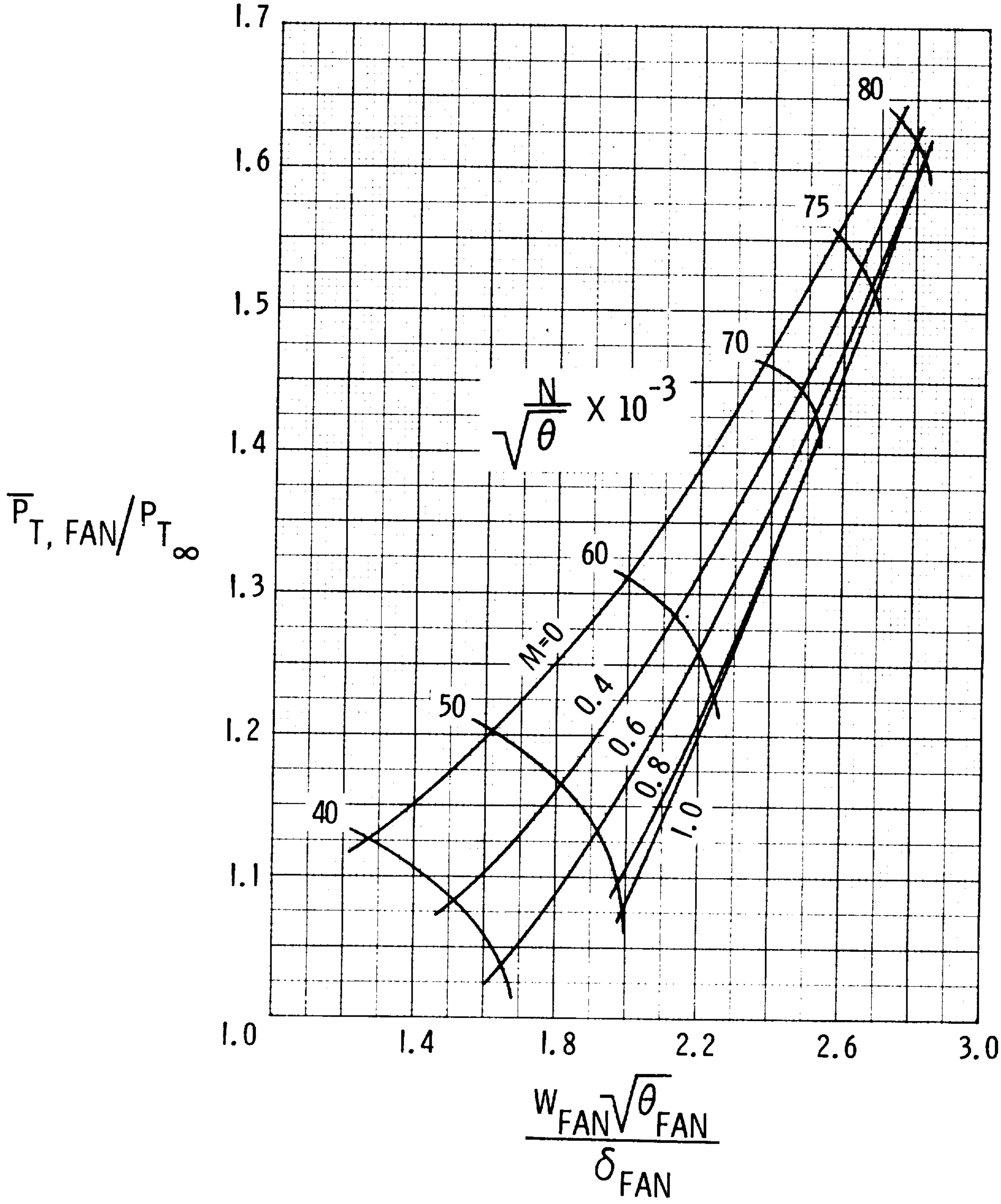


Figure IV-20. Fan corrected weight flow map.

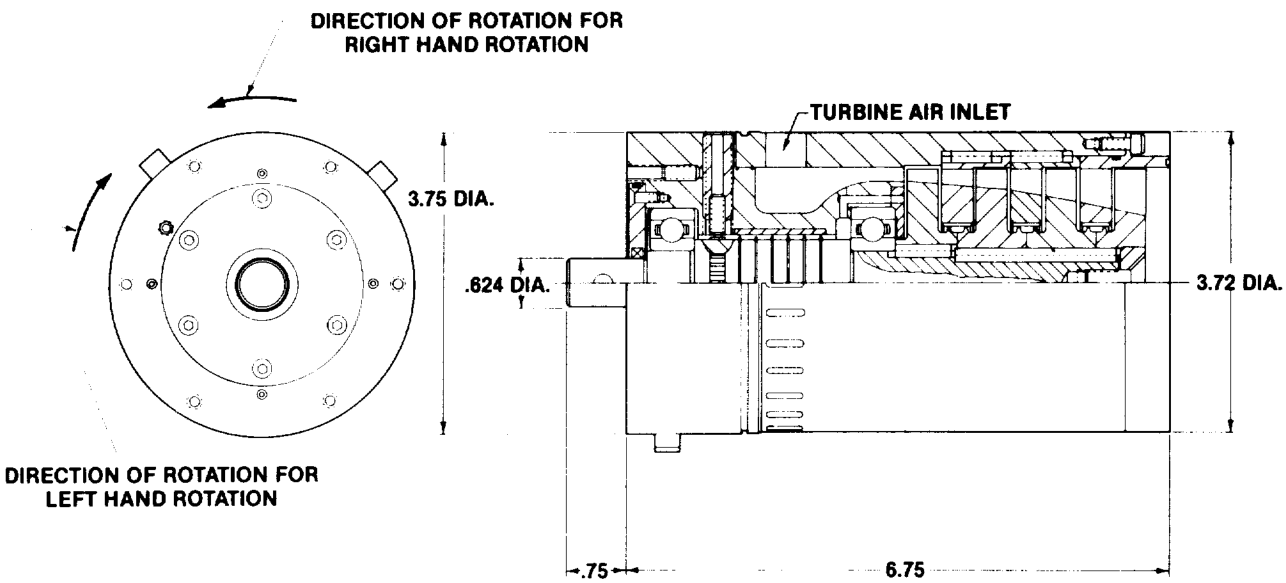


Figure IV-21. Model 666H Single rotation air turbine motor. (All dimensions are in inches.)

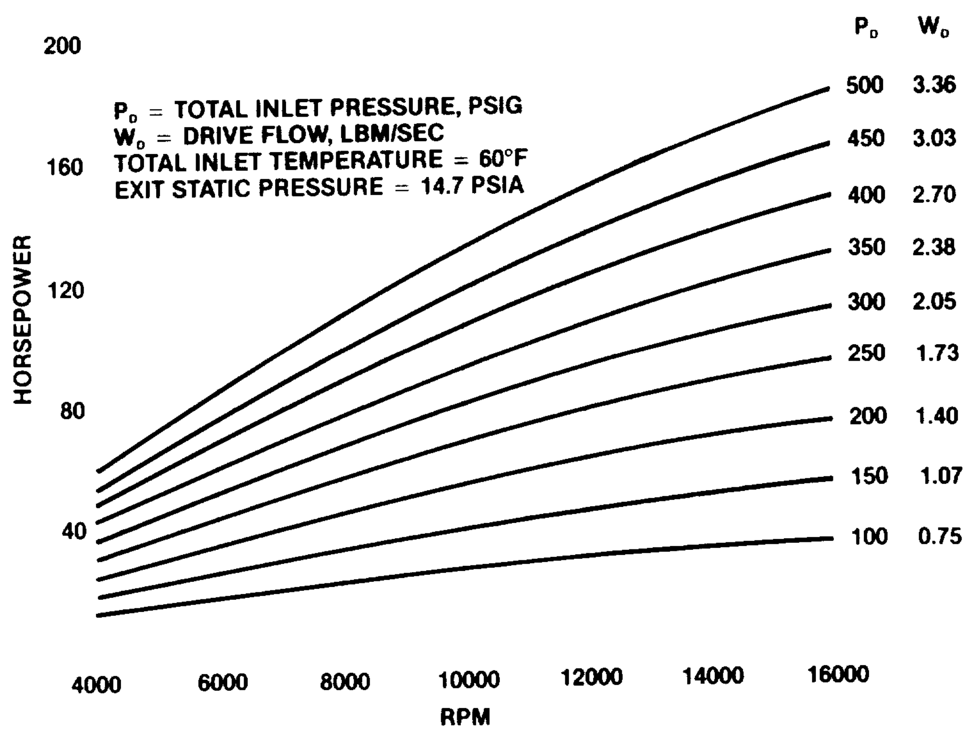


Figure IV-22. Performance curves for Model 666H motor.

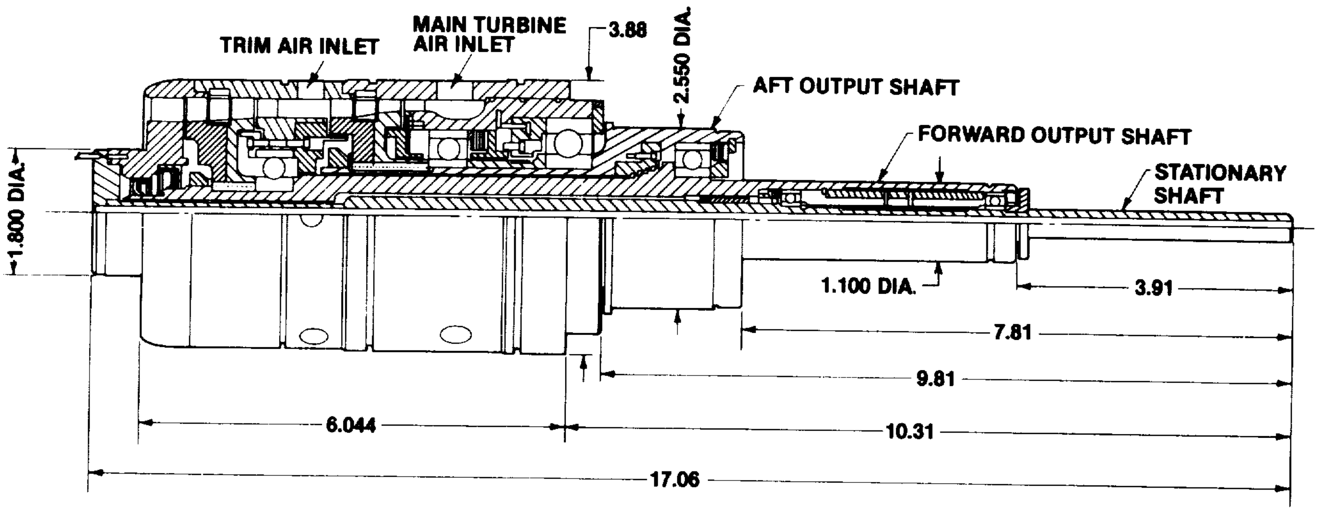


Figure IV-23. Model 1700B Counter-rotating air turbine motor. (All dimensions are in inches.)

LEGEND

CURVE NUMBER	1	2	3	4	5
INLET TOTAL PRESSURE, PSIA	700	600	500	400	300
INLET TOTAL TEMPERATURE, °F	100	100	100	100	100
EXIT STATIC PRESSURE, PSIA	215.7	184.9	154.1	123.3	92.4
TOTAL FLOW, LBM/SEC	6.26	5.37	4.47	3.57	2.68

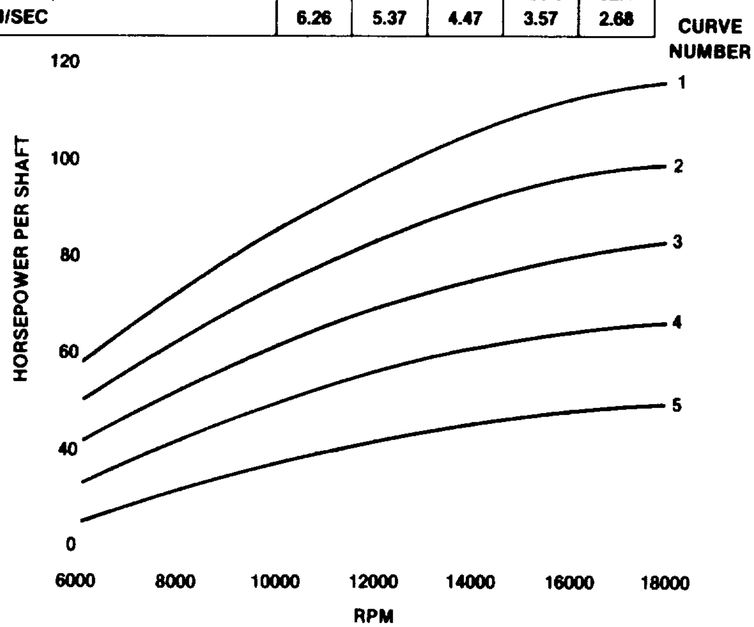


Figure IV-24. Performance curves for Model 1700B motor.

SECTION V - Calibration of Propulsion Simulation Systems

Prior to a wind-tunnel investigation, a series of calibrations are performed in order to determine mass flow characteristics, jet-off balance/bellows interactions and momentum tares. In general the following are performed:

1. Determine effective discharge coefficient of sonic nozzle metering devices, if used.
2. Determine jet-off balance/bellows tare forces
3. Determine momentum tare forces

Prior to 1990, calibrations of propulsion simulation systems were generally conducted in the tunnel test section immediately preceding the wind-on testing. These calibrations are time consuming, requiring as much as 2 weeks of tunnel down time. It has long been recognized as an inefficient method of operation, so during the period from mid 1989 to 1990, while the tunnel was shut down for major modifications to the support system, a model preparation room was constructed. It is envisioned that all pretest propulsion simulation calibrations will be conducted in this room prior to tunnel entry, thereby avoiding lengthy delays in the tunnel run schedule. A description of the model preparation area will be provided later in this section.

A. Calibration Nozzles.- Three sets of calibration nozzles are available for use with the various propulsion simulation systems used at the 16-Foot Transonic Tunnel. For each system, a range of nozzle throat areas and perforated plates of varying porosity is available. Because both bellows pressure and mass flow can have an effect on momentum tares, and because the internal chamber pressure for a constant mass flow rate is in practice somewhat dependent upon open area at the perforated plate and nozzle it is advisable, wherever possible, to perform the blowing calibrations with nozzles and perforated plates having the same or similar geometric characteristics to test hardware. Figure V-1 presents details of the calibration nozzles for the single-engine system and figures V-2 and V-3 show the nozzles for the twin-engine systems.

B. Multiple Critical Venturi System.- Because the accurate measurement of air weight flow is critical to obtaining accurate propulsion-model data, the primary mass flow measurement system in the 16-Foot Transonic Tunnel is a MCV system. This multiple venturi system has been calibrated against another MCV system whose calibration was certified by the Colorado Engineering Experiment Station, Inc (CEESI) to have measurement uncertainty within 0.07 percent over an airflow range from 0.1 lb/sec to 20.0 lb/sec. Additional details of this MCV system are provided in SectionVI-D.

C. Chamber Flow Rate.- The eight sonic nozzles or orifices located within the flow transfer assemblies (bellows) can be used as an alternative to the Multiple-Critical Venturi System (or other venturi systems) for measuring the mass flow rate by obtaining an effective orifice discharge coefficient from the calibration nozzle tests. The orifices act as critical nozzles since they are always choked. The ideal flow rate through a choked nozzle is:

$$\dot{m}_{c,i} = A_o \frac{P_c}{\sqrt{T_c}} \sqrt{\frac{\gamma}{R}\left(\frac{2}{\gamma+1}\right)^{\frac{\gamma+1}{\gamma-1}}} \quad \text{slugs/sec}$$

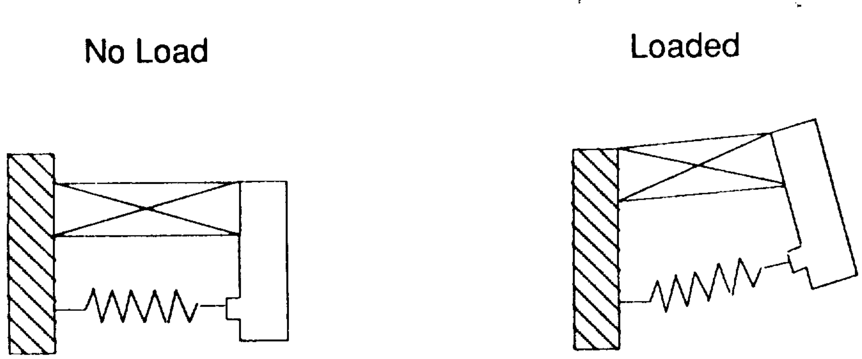
Known values during the test are: chamber pressure, P_c; chamber temperature, T_c; and the total geometric orifice area A_o. From continuity relations, the flow rate through the orifices is equal to the flow through the calibration nozzles, so that the effective discharge coefficient for the orifice can be determined from:

$$c_{d,c} = \frac{\dot{m}_a}{\dot{m}_{c,i}} = \frac{.995\dot{m}_i}{\dot{m}_{c,i}} = \frac{\dot{m}_c}{\dot{m}_{c,i}}$$

The orifice discharge coefficient increases slightly with chamber pressure so that it generally may be expressed in the form; $C_{d,c} = I_c + K_c P_c$, where I_c is the intercept and K_c is the slope with increasing chamber pressure. This increase in discharge coefficient with pressure ratio is a characteristic that has been observed for thick plate orifices. Again, by continuity, the ratio of chamber pressure to calibration nozzle total pressure when both outlets are choked must be:

$$\frac{p_c}{p_{t,j}} = \frac{A_t}{A_o}\frac{T_c}{T_{t,j}}\frac{0.995}{c_{d,c}}$$

D. Jet-Off Balance/Bellows Tares.- Jet-off force and moment interactions (or tares) exist between the bellows/flow transfer system and force balances. These tares have been found for both single and twin-jet bellows systems and are a result, as shown schematically in the following sketch, of the bellows acting as a spring connecting the metric portion of the model to ground.



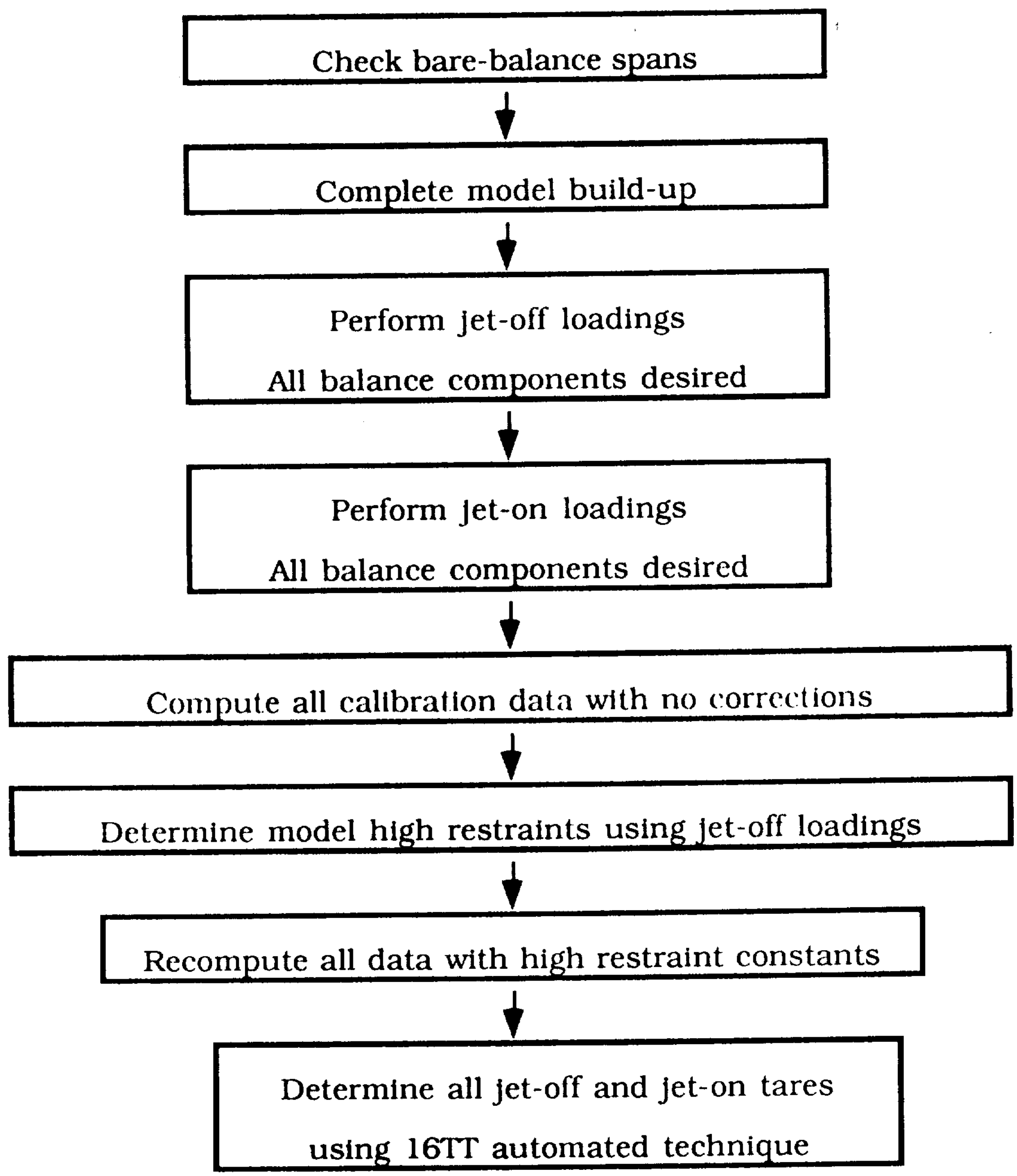
Consequently, single and combined calibration loadings of all required forces and moments are made with completely assembled models. The desired result of these calibrations is to account for not only the bellows/balance interactions, but any other restraints across the balance such as pressure tubing, instrumentation cables, etc. Note that these interactions are in addition to the bare balance interaction corrections usually made.

Provision for calibration fixtures should be part of any powered model design. These fixtures should be compatible with the loading capabilities available in the model preparation area and 16-Foot Transonic Tunnel.

E. Momentum (Flow) Tares.- Single and combined loadings are also applied to the model at jet-on conditions with the calibration nozzles. These calibrations are performed with the jets operating because this condition gives a more realistic effect of pressurizing the bellows than does capping off the nozzles and pressurizing the flow system. Loadings can also be done in the axial-force direction with the flow system capped off and pressurized. However, this method generally is less accurate than the loading/blowing method. It should be noted that these loadings are done in conjunction with the jet-off loadings.

The momentum tare forces are a function of the combination of average bellows internal pressure and mass flow through the system. Although the bellows were designed to minimize momentum and pressurization tares, small bellows tares still exist with the jet on. These tares result from small pressure differences between the ends of the bellows when internal velocities are high, small differences in the forward and aft bellows spring constants when the bellows are pressurized, and small misalignments in force vectors as mass flow exits each of the sonic nozzles.

F. <u>Calibration Procedure</u>.- The following flow chart is included to illustrate the calibration procedure. The 16-Foot Tunnel data reduction codes include tare equations that are used for the propulsion simulation systems described. These equations are very general, however, and can be modified to fit a particular set of requirements.



G. <u>Typical Tare Equations</u>. - The following are examples of the tare equations for axial force and for normal force. These two represent examples of the set of equations currently programmed as part of the standard 16-Foot Transonic Tunnel data reduction program. The complexity of these equations has evolved over a number of years, along with the propulsion testing requirements. It is likely that they will change with further advances in propulsion test technique.

$$
\begin{aligned}
TAREN = \; & xk(1,I)+xk(2,I)*NF+xk(3,I)*PM+xk(4,I)*RM+xk(5,I)*YM+ \\
& xk(6,I)*SF+ \; DELP\{xk(46,I)+xk(47,I)*NF+xk(48,I)*PM+ \\
& xk(49,I)*RM+xk(50,I)*YM+xk(51,I)*SF+PL1*xk(52,I)+ \\
& PL1^2*xk(53,I)+PL2*xk(54,I)+PL2^2*xk(55,I)+AREA[xk(56,I) \\
& +xk(57,I)*NF+xk(58,I)*PM+xk(59,I)*RM+xk(60,I)*YM+ \\
& xk(61,I)*SF]+AREA^2[xk(62,I)+xk(63,I)*NF+xk(64,I)*PM+ \\
& xk(65,I)*RM+xk(66,I)*YM+xk(67,I)*SF]\}
\end{aligned}
$$

$$
\begin{aligned}
TAREA = \; & xk(7,I)+xk(8,I)*NF+xk(9,I)*PM+xk(10,I)*RM+ \\
& xk(11,I)*YM+xk(12,I)*SF+xk(37,I)+xk(38,I)*DELP+ \\
& xk(39,I)*DELP^2+AREA[xk(40,I)+xk(41,I)*DELP+ \\
& xk(42,I)DELP^2]+AREA^2[xk(43,I)+xk(44,I)*DELP+ \\
& xk(45,I)*DELP^2]
\end{aligned}
$$

In the above equations: NF is normal force, PM is pitching moment, RM is rolling moment, YM is yawing moment, and SF is side force, AREA is typically the nozzle area, DELP represents the pressure differential between the bellows pressure and ambient pressure in the model cavity, PL1 and PL2 represent model preload variables, I represents balance number (up to 3 balances allowed), and xk represents the various tare constants. Equations similar to the TAREN equation above are available for correcting pitching, rolling, and yawing moment and side force. In all, there are 155 constants (per balance) available for applying the bellows/balance and momentum tare interactions.

H. Model Preparation Area- The model preparation area allows pretest buildup and calibration of propulsion models. The room consists of a calibration bay area with a 10'8" high ceiling and a control room as shown in figure V-4. The area is completely enclosed providing physical security for most test requirements. Duplicate instrumentation, data acquisition, data reduction, and air supply systems provide the same capabilities for complete model calibrations as provided in the 16-Foot Transonic Tunnel. Because of the relatively small test bay room size, exhaust flow must be vented in an effort to both reduce recirculation of this exhaust flow and prevent large variations in ambient pressure. Exhaust flow is, therefore, directed toward a 5-foot by 5-foot exhaust vent, which ducts the flow outside of the bay area. Test bay walls are acoustically treated for noise reduction purposes.

A model support system shown in figure V-5, has an attachment station which is identical to that of the tunnel, accepting any of the standard butts shown in figure II-5. The support system does not have a remote roll capability, however, can be manually rolled when desired. A pitch drive system allows for ± 5° variation in angle of attack, allowing most models the ability to be leveled while loading during calibrations. The support system is designed for loading to 10,000 lbs normal force and 15,000 in-lbs pitching moment (about a point 95.25 inches downstream of the butt attachment) with a model/sting combination weighing 1500 lb.

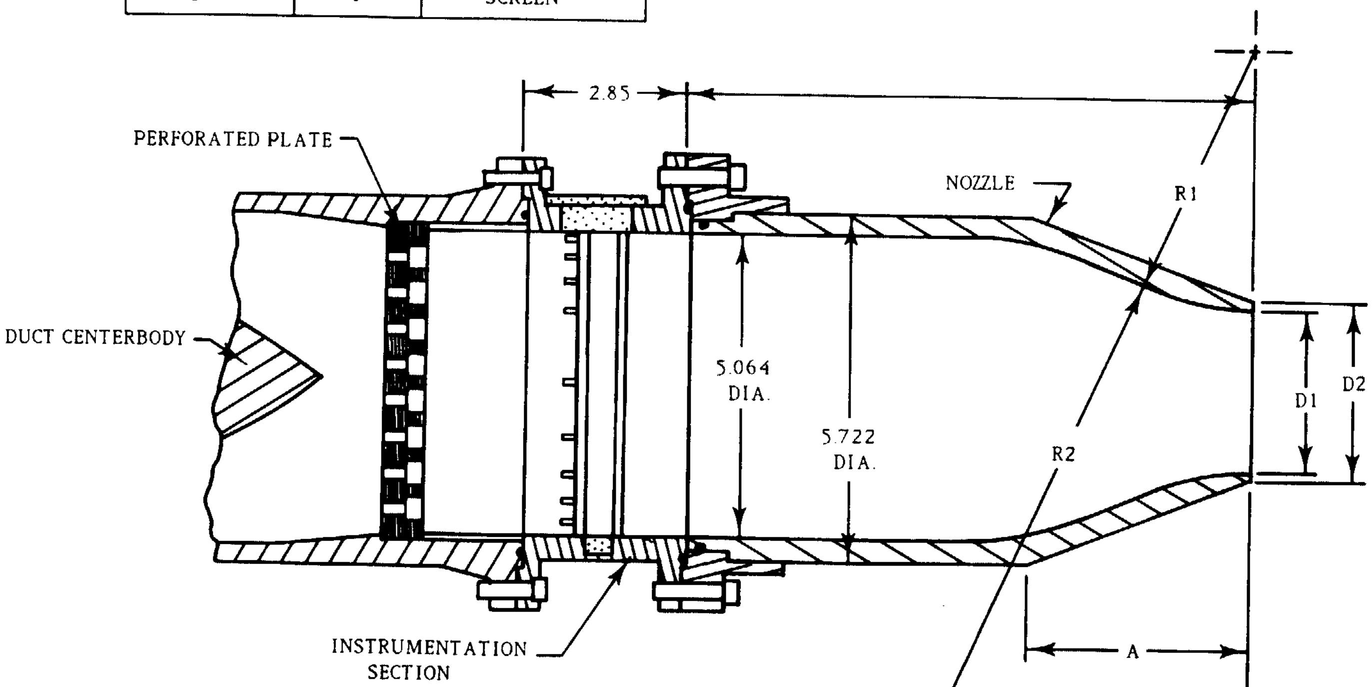
Dual 1800 psi air supply systems can provide independently controlled 15 lb/sec weight flows (each) to the model, however, at present these are the same systems supplying the 16-Foot Transonic Tunnel. As a result, some scheduling conflicts are possible, especially if both 1800 psi air supply systems are needed in the tunnel. However, these conflicts can normally be handled by off shift operation in the model preparation area.

Also included in the calibration bay area is a model load calibration platform. A sketch of this calibration platform is provided in figure V-6. The calibration beam is capable of both longitudinal and lateral adjustment to provide precise loading capability. The calibration beam can move up to ±12 inches laterally and ±24 inches in the axial direction. The platform was designed to handle up to a 10,000 lbf vertical calibration force. This along with the ability to roll and level the model should allow for complete calibration flexibility for most model installations.

Once pretest calibrations with the fully assembled model are complete, the model must be disconnected from the model preparation area and moved into the 16-Foot Transonic Tunnel test section. A model support cart has been fabricated to facilitate in the transportation and placement of the fully assembled model in the tunnel test section. A sketch of this model support cart is provided in figure V-7. While this sketch is not completely accurate, it does serve to illustrate some of the key features. The cart has a hydraulic lift system which can position the cart table surface from 20 inches off the floor (when fully collapsed) to approximately 106 inches off the floor when fully extended. This range of movement is required to handle the removal of the assembled model from the model preparation area and subsequent installation of the model on the test section centerline. The model-sting support assembly (fixture which actually holds the model and sting assembly) rests on a fixture which is fully adjustable in both the axial and lateral directions to allow for optimum location of the model and model support for lifting into the tunnel test section and mating with the tunnel support system hardware. (This positioning hardware is not shown in the figure). A sketch of the model-sting support assembly hardware is shown in figure V-8. This hardware will typically be test specific, hence arrangements for correctly sized sting insert clamps must be made prior to model arrival at the 16-Foot Transonic Tunnel Complex.

PERFORATED PLATES AVAILABLE		
OPEN AREA (sq.in)	% DUCT AREA	PERFORATION SIZE (DRILL NO.)
2.460	12.2	32
2.870	14.2	30
3.853	19.1	26
5.649	28.0	15
7.463	37.1	5
15.286	75.9	SCREEN

NO.	STRATFORD CHOKE NOZZLES AVAILABLE						
	THROAT AREA (sq.in.)	DESIGN GEOMETRY					
		R1, in.	R2, in.	X, in	D1, in.	D2, in	A, in
6	1.000	2.257	21.31	9.428	1.128	1.378	5.50
1	1.936	3.140	9.000	6.274	1.570	1.820	4.00
7	3.000	*	*	*	1.954	*	*
4	3.992	4.510	8.320	5.837	2.255	2.505	4.00
2	5.711	5.400	7.700	5.432	2.700	2.950	4.00
5	8.501	6.580	7.868	4.985	3.290	3.540	4.00
3	11.352	7.600	5.900	4.086	3.800	4.050	3.50



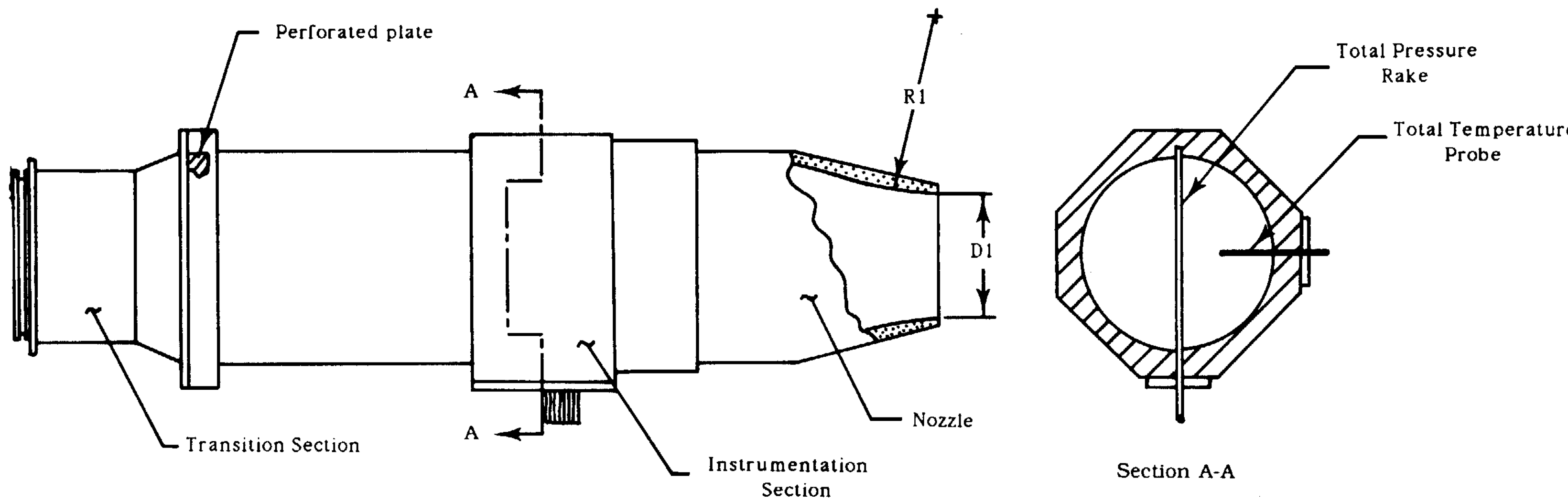
NOZZLES 1, 2, AND 3--DRAWING LC-510436
NOZZLES 4 AND 5--DRAWING LC-904400
NOZZLE 6--DRAWING LC-943527
* NOZZLE 7--ORIGINAL DRAWING LOST
MATERIAL: 2024-T4 ALUMINUM

INSTRUMENTATION SECTION--DRAWING LE-1071906A
MATERIAL: 7075-T6 ALUMINUM

Figure V-1. Sketch of calibration nozzles and associated hardware used with single-engine simulator. (All dimensions are in inches.)

Perforated Plates Available	
Open area, sq. in.	Percent open
1.925 3.657 8.136	18.9 35.9 80.0

Throat area/nozzle sq. in.	Nozzle type	D1, in	R1, in	Drawing
2.201 3.692 5.416	Stratford choke ↓	1.674 2.168 2.626	3.348 4.336 5.252	LC-904420 ↓
4.680	ASME	2.441	4.882	GAC 399MOD1031A
2.835 4.680 6.505	Iris convergent ↓	1.900 2.441 2.878		GAC 399MOD1011A ↓



NOTES:

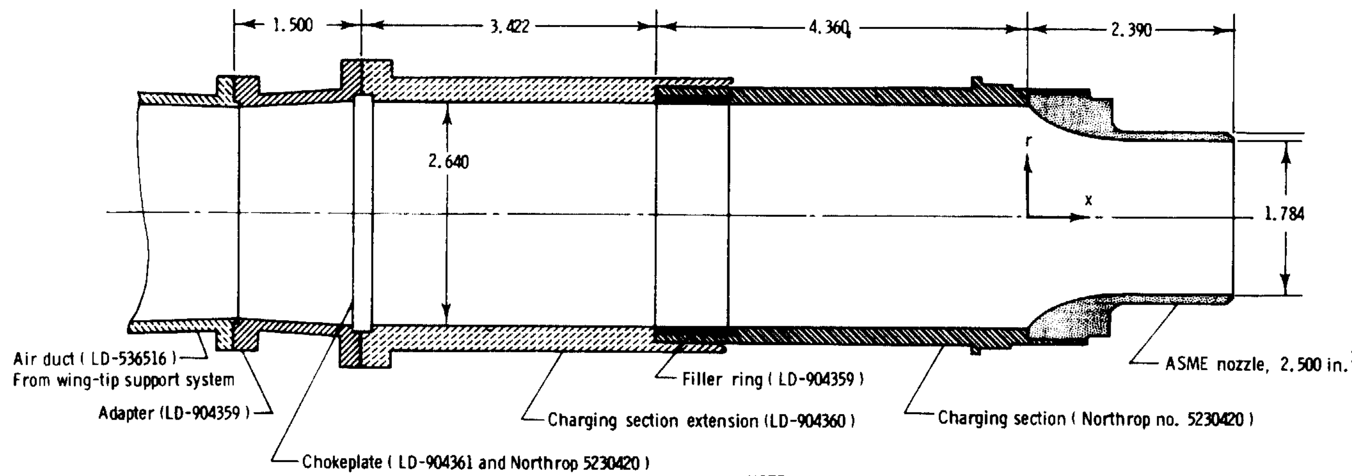
A. System leak check plug, drawing LC-904421 (100 psi max).

B. Perforated plates, drawings LC-904408 and LB-904434.

C. Adaptor LC-904407 can be attached to air ducts LD-536516 of the wing tip support system.

D. Alternate assemblies and hardware, drawing LB-904438 (Assembly)
   Rakes, drawing LC-904432A
   Thermocouple, drawing LB-904433A
   Instrumentation section, drawing LD-904430A
   High load system adaptor, drawing LC-904436
   Grumman diffuser adaptor, drawing LC-904437

Figure V-2. Sketch of calibration nozzles and associated hardware used with twin-engine simulators. System 1. (All dimensions are in inches.)



x, in	r, in
0.000	1.320
.010	1.270
.037	1.220
.084	1.170
.154	1.120
.249	1.070
.380	1.020
.562	.970
.852	.920
1.320	.892

CHOKEPLATES AVAILABLE	
Open area, in^2	Percent open
0.9800	17.9
1.3146	24.0
1.6677	30.0

NOTE:

A. Reference drawings
   1. Assembly, LD-904358
   2. Northrop drawing 5230420

B. Exit plug (LD-904361) available to leak check entire system. Maximum operating pressure is 250 psi.

C. Charging section and nozzle can be attached to twin-jet system.

Figure V-3. Sketch of calibration nozzle and associated hardware used with twin-engine simulators. System 2. (All dimensions are in inches.)

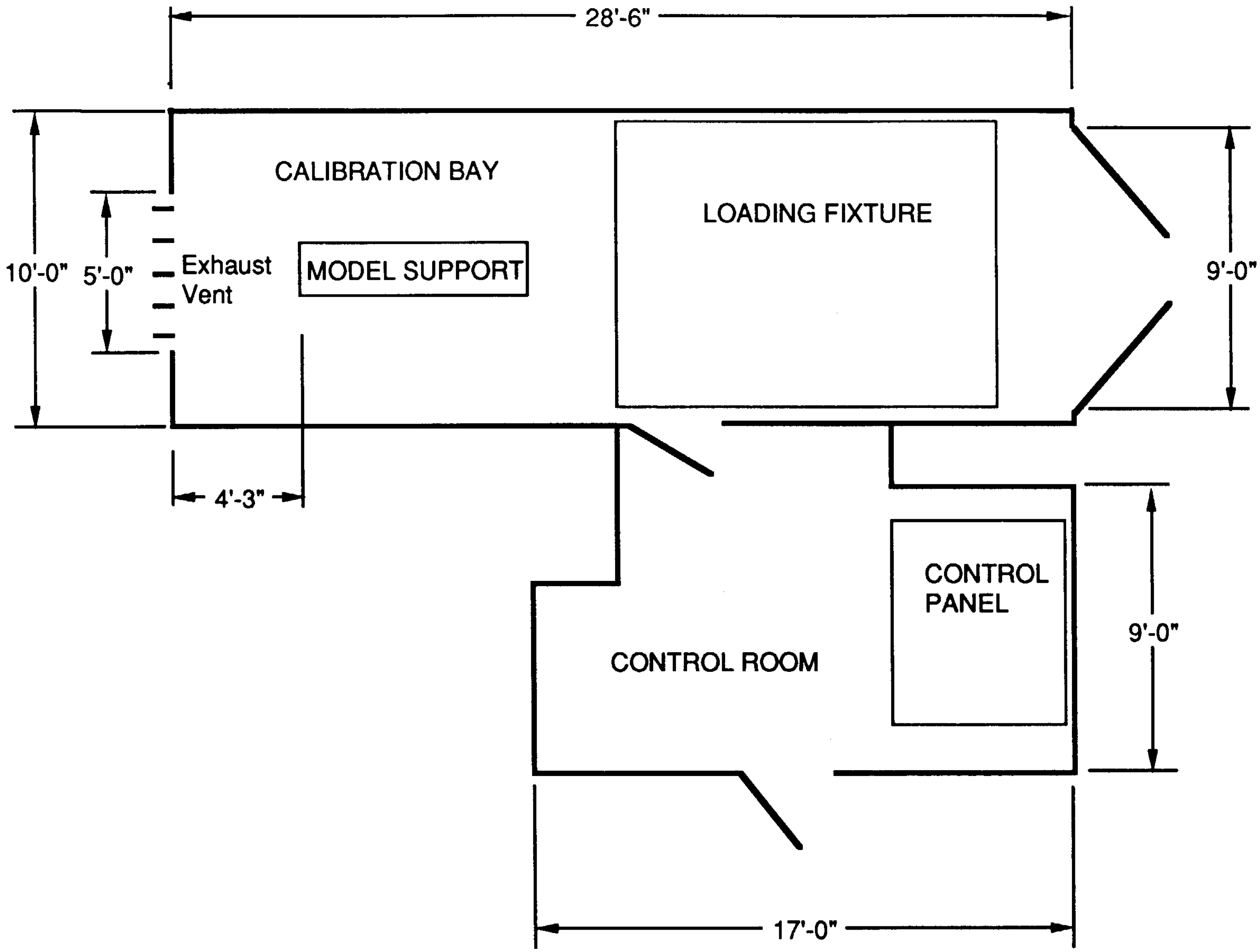


Figure V-4. Layout of Model Preparation Area.

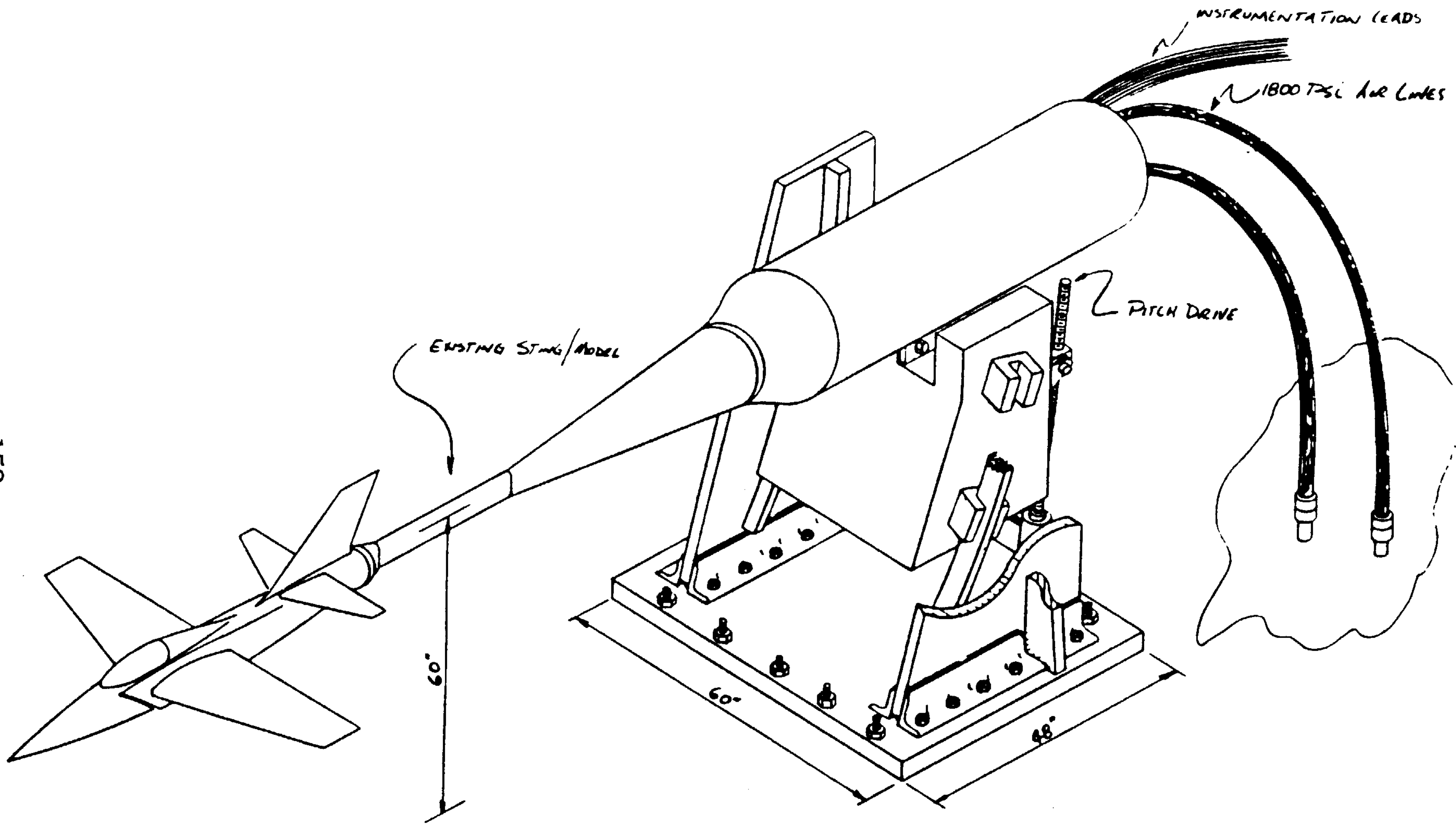


Figure V-5. Sketch of model preparation area model support system.

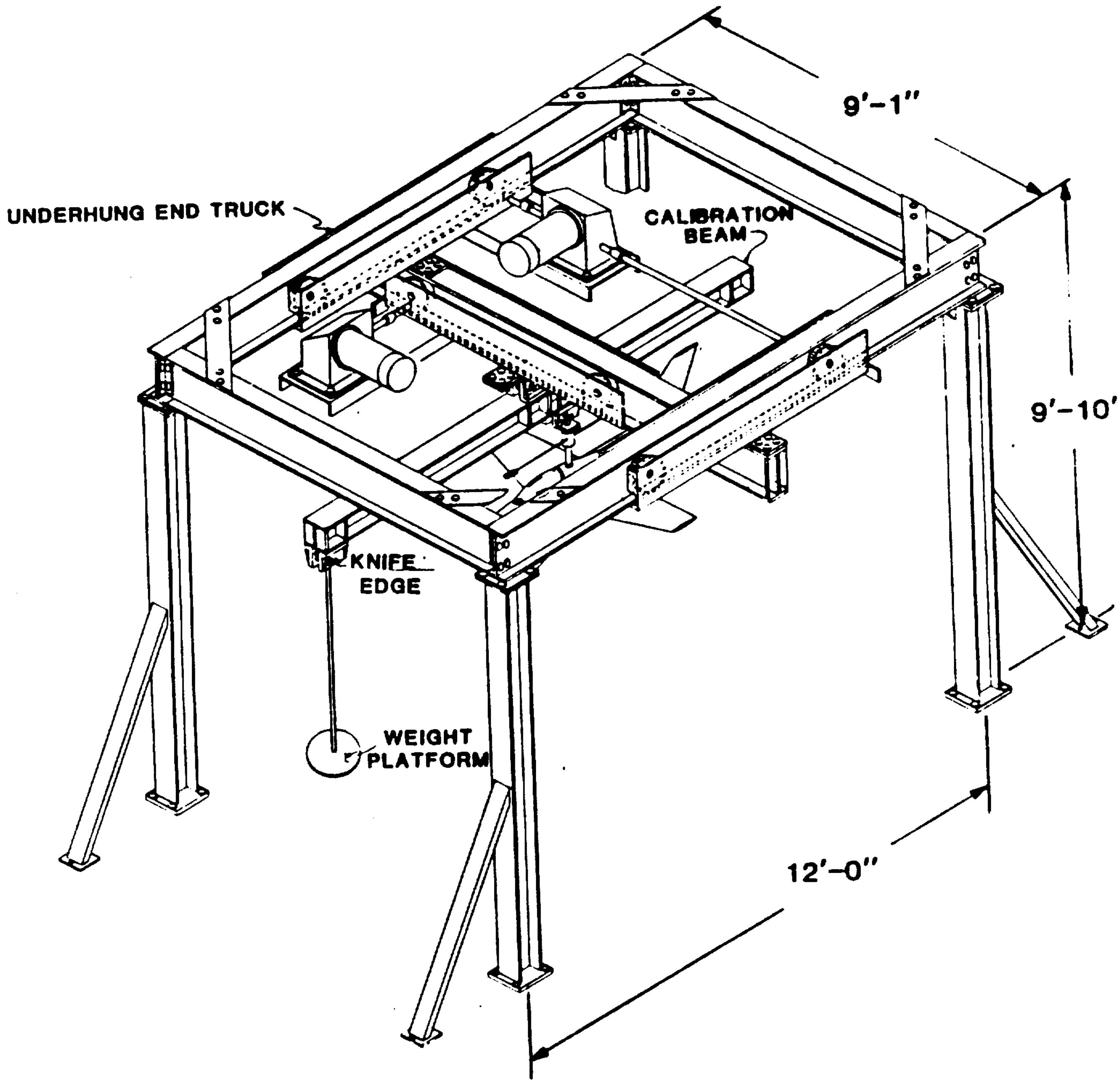


Figure V-6. Sketch of model load calibration platform.

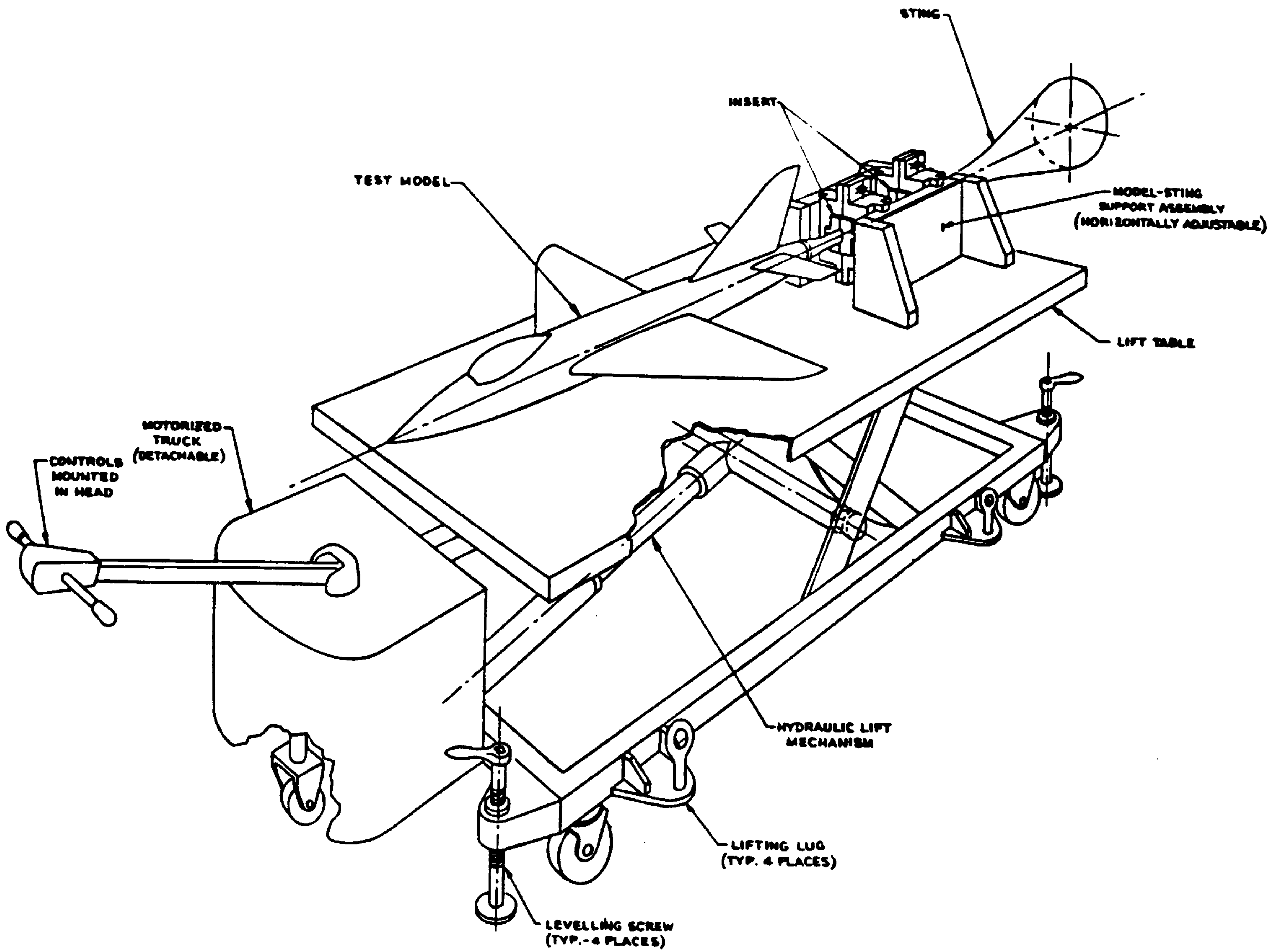
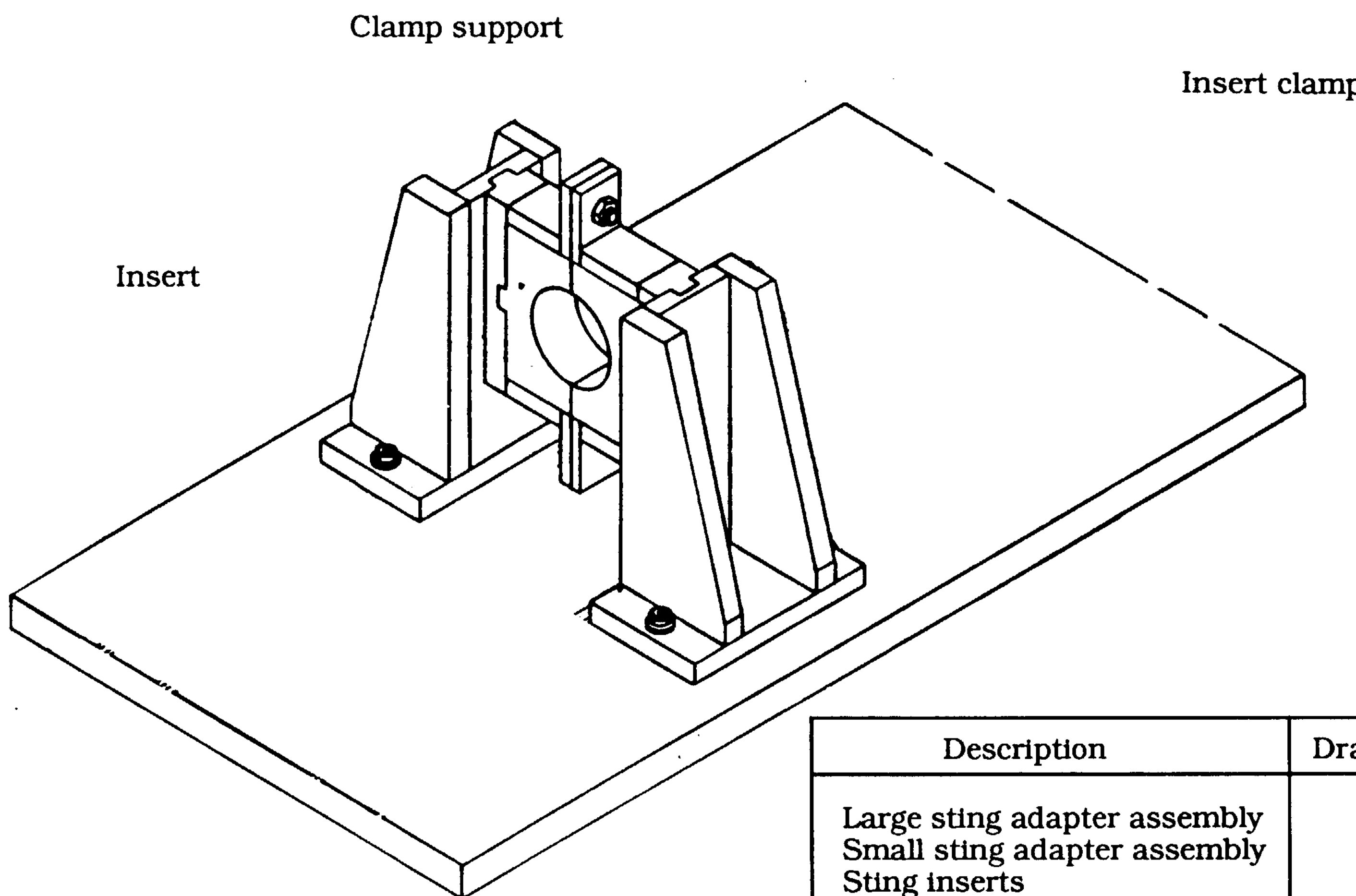


Figure V-7. Sketch of model support cart.



Description	Drawing number
Large sting adapter assembly	LD-544490
Small sting adapter assembly	LD-544491
Sting inserts	LD-944492
Sting insert clamps	LD-944493
Large sting clamp support	LD-944494
Small sting clamp support	LD-944495
Base plate	LD-944496

Figure V-8. Sketch of model-sting support assembly hardware for use with the model support cart.

SECTION VI - Instrumentation

A. Force and Moment Measurements.- Model force and moment measurements in the Langley 16-Foot Transonic Tunnel and Static Test Facility are made with internal strain gage balances. Sketches of and load ranges for the principal strain gage balances used in this facility for propulsion aerodynamics testing are presented in figures VI-1 to VI-12. The table below indicates the strain gage balances normally used with the various propulsion simulation systems that are available and the pertinent figures.

Model System	Balance	Figure
Single Engine	711A & B	VI-1
	1614A & B	VI-3
	1631 & 1631B	VI-8
Twin Engine	1621A & B	VI-5
	1630B	VI-7
Wing-tip Support	1617	VI-4
	1621A & B	VI-5
	1630B	VI-7
Dual-Flow	1635	VI-10
Semi-Span	804SB, SC, SA-M	VI-2
Turboprop Simulator	1633CRA & CRF	VI-9
Turboprop Thrust-Drag	1617	VI-4
	1629	VI-6
Flow-through	1627	VI-11-12

Many other Langley balances are available with listing and description of these balances provided to the user upon request. Availability of Langley balances must be negotiated early in test planning. Force/moment balances may also be supplied by users. A user-supplied balance must be available for bench calibration at Langley before test entry. Normal bench calibration of force balances requires a minimum of 5 weeks. Therefore, a user-supplied strain gage balance must be delivered to Langley at least 6 weeks ahead of the scheduled build-up of the model.

Figure VI-11 is a sketch of the six-component, flow-through 1627 balance; the balance to sting adapter; and the balance to transport model adapter to show the important dimensions of the assembly and details of the balance construction. This balance is identified separately because it is a flow through type balance, having its own internal bellows systems, and, therefore, is a very unique type of balance. Information on the sting used with this balance can be found in Section III (Model Support Systems), and on drawing number LE-534599. Total thrust-minus-drag data from the balance are combined with thrust data obtained by pretest calibration and internal flow measurements from tunnel runs to generate power-on external drag polars. The maximum design loads for this balance are as follows:

Normal Force	3,000 lbs.
Axial Force	500 lbs.
Pitching Moment	15,000 in-lbs.
Rolling Moment	12,000 in-lbs.
Yawing Moment	15,000 in-lbs.
Side Force	3,000 lbs.

Figure VI-12 is a photograph of the 1627 balance (background) and the extra air supply tube/bellows assembly (foreground). For high-pressure-air powered models with faired over inlets a maximum of approximately 17 lb/sec mass flow can be obtained through the sting/balance system when installed in the 16-Foot Transonic Tunnel. Instrumentation requirements for the balance include the usual balance leads (not shown in the photograph since they are connected using a 32 pin miniplug) and two bellows cavity pressures feeding through the aft balance flange.

B. Pressure Measurements.- 16-Foot Transonic Tunnel freestream stagnation pressure and tunnel tank static pressure are measured with Ruska absolute pressure sensors which have 0.018 percent accuracy. A backup measurement system for freestream stagnation pressure is a sonar manometer, and a backup measurement for tank static pressure is a Digiquartz absolute pressure sensor. In addition, a second backup system is available which uses differential pressure transducers. Atmospheric pressure is measured with a 15 psi absolute pressure transducer.

Individual differential or absolute pressure transducers available for measuring model pressures range from 0.5 psi to 2000 psi. Up to sixteen Electronically Scanned Pressure (ESP) modules may be used in a model. These ESP modules, shown in figure VI-13, have 32 pressure ports per module and are available in the following sizes: 5 psi, 10 psi, 15 psi, 45 psi, and 100 psi. A rack-mounted (external to the model) ESP system containing two 100 psi and two 250 psi modules with 32 ports each is also available.

C. Temperature Measurements.- 16-Foot Transonic Tunnel freestream stagnation temperature is measured with up to four platinum resistance thermometers. The wind tunnel dew point temperature is measured with a General Eastern hygrometer (model M2). Individual thermocouples are used to make all other temperature measurements.

Chromel-Alumel, Iron-Constantan, and Copper-Constantan thermocouples are preferred. A 32°F cold junction reference box is available for use with these thermocouples.

D. Weight Flow Rate Measurements.- Weight flow rate of the high-pressure air used for jet exhaust simulation in the 16-Foot Transonic Tunnel is measured by a multiple critical venturi system shown in figure VI-14. This system provides for high accuracy of flow measurement, an extremely wide range of weight flow, small pressure losses, and a very low level of noise in the airstream and pipe structures. The system was calibrated in the Boeing Airflow Calibration Facility over a pressure range from approximately 36 psi to 920 psi and has a calibration accuracy of 0.1 percent. The six critical-flow venturies (fig VI-14 (c)) are of varying size, each with its own screw-on cap (with O-ring seal to prevent leakage) so that 47 increments of flow area are available to meet test requirements. As shown in figure VI-15, the system provides a weight-flow range from 0.014 lb/sec at 20 psi to 44 lb/sec at 1500 psi. More information can be found in Reference VI-1.

The Static Test Facility, which also uses high-pressure air for jet exhaust simulation, measures weight flow rate with two critical flow venturis. Secondary weight flow rate in both the 16-Foot Transonic Tunnel and the Static Test Facility are measured with a turbine flowmeter. Weight flow rate in either facility can also be determined by measurement of the pressure and temperature of the flow through sonic nozzles located inside the various propulsion simulation systems.

E. Flow Field Measurements.- Three systems are available in the Langley 16-Foot Transonic Tunnel for making measurements of the flow surrounding models in the tunnel test section. Both remotely and manually controlled translating survey mechanisms are available for making local pressure and temperature measurements and a laser-velocimeter system mounted in the test section plenum is available for making local velocity measurements.

A sketch showing the various components of the remotely controlled translating survey mechanism is presented as figure VI-16. A probe mounted on this survey mechanism can be remotely positioned anywhere in a cylindrical survey volume that is approximately 4 feet in diameter and 4 feet long. The existing device supports are not compatible with the new tunnel strut head, therefore plans for fabrication of additional hardware must be made well in advance of a tunnel entry if the survey mechanism is to be employed.

The manual translating sting mechanism is attached to the strut head through the use of a 16-Foot Tunnel standard butt as shown in figure VI-17. The 60 inch rake holder is capable of holding 9 static, total, or dynamic pressure probes depending upon test requirements (of course, other rake holders can be made). Using the remote roll capability of the support system, the existing rake is capable of surveying a cylindrical volume that is approximately 5 feet in diameter and 6.5 feet long. Because of the method of attachment, this system will be used with the semi-span support system, and during tunnel flowfield calibrations only.

The 16-Foot Transonic Tunnel laser velocimeter system is a non-intrusive flow measurement device which allows the determination of two components of the flow velocity (streamwise and vertical) at a point in the tunnel test section. The laser and associated optics are mounted on a moveable scanning rig located in the test section plenum (see figure VI-17) which allows the laser to survey a volume of the test section. The surveyed volume is limited by the size of the window in the test section wall and the transmitting lens size. The volume which can be surveyed using the scan rig encompasses streamwise + 21 1/2 inches from Tunnel Station 134 ft. 4 in. and 14 1/2 inches above and below the tunnel center line. A photograph of a semi-hemispherical model test setup is shown in figure VI-18. The spanwise movement of the sample volume is obtained by use of a zoom lens which allows a movement of + 39 inches about the tunnel center line. The spatial resolution of the system is such that the sample volume is a cylinder with a diameter of 0.012 inches and a length of 0.24 inches. Data are acquired and recorded on magnetic or optical disk by a computer dedicated to the laser velocimeter. In addition to acquisition and recording, the computer reduces the data such that the velocities are obtained almost "real time" and are displayed on a CRT.

The laser velocimeter relies on the Doppler Shift in the laser light reflected from particles moving with the flow to determine the flow velocities. In order to obtain maximum accuracy (designed to be 1 percent) over the entire Mach number range, the particles must be small enough to follow the flow accurately even in areas of high velocity gradients. In order to insure an adequate supply of small particles, the 16-Foot Transonic Tunnel is seeded upstream of the test section with polystyrene microspheres with a diameter of 0.8 micron. The seeding system consists of an array of particle generators mounted on a remotely controllable seeding rig in the settling chamber upstream of the anti-turbulence screen so that there is no effect of the array on the test section flow quality.

F. <u>Flow Visualization Equipment</u>.- The 16-Foot Transonic Tunnel has the capability for model flow diagnosis by use of several flow visualization techniques. Included in the techniques available are fluorescent oil flow, oil flow, "ink" flow, "permanent record" oil flow, and laser light sheet. In the fluorescent oil flow technique, an oil which fluoresces under ultraviolet light is painted on the model prior to the tunnel run, the model is illuminated with ultraviolet light during the run, and the flow patterns shown by the fluorescing oil photographed. In the oil flow technique a mixture of oil and lamp black is spotted at strategic points on the model (generally painted white) prior to the run and the flow patterns during the run are photographed and/or videotaped with white light illumination. In the "ink" flow technique, a mixture of water and any one of a number of coloring agents ("ink") is pumped out of a series of orifices installed in the model (also generally painted white) upstream of the area of interest during the run and the resultant flow patterns are photographed and videotaped with white light. In the permanent record oil flow technique, a mixture of linseed oil based paint, tempura paint, and vacuum pump oil is applied to the model surface of interest and then the tunnel is run at the desired condition until the mixture has dried on the model surface. This provides a permanent record of the surface flow which can then be photographed in detail at a later time. The laser light sheet can be used to examine flow features off of the model surface. For this process, the tunnel flow is seeded with water vapor using the LDV seeding supply rig. The light sheet flow visualization can be recorded on video tape.

Still photography is accomplished for the flow visualization studies with up to four Hasselblad single-lens reflex cameras. These cameras use 70 mm roll film (generally Kodak Tri-X Pan ASA 400 although other film can be obtained with proper notice) in an approximate 70 shot magazine. These cameras are motor driven with remote actuation which allows them to be mounted at various locations in the test section plenum chamber. Lenses available for the Hasselblads are 2-80 mm, 1-125 mm, 1-150 mm, and 1-250 mm.

Video cameras are available and are permanently installed in the test section plenum chamber. One camera is mounted behind the test section window and views the right-hand side of the model. The other camera is mounted in the top of the test section looking down through a "port-hole" at the top of the model. The side TV camera has the capability to be remotely panned, tilted, zoomed, etc. while the top camera has somewhat less flexibility due to the small window it must look through. Either one or both cameras can be connected to videotape recorders in the tunnel control room. Motion picture cameras have been used in the tunnel at times in the past, but at present videotape is the preferred method for continuous recording.

Continuous white light illumination of the model in the tunnel is provided by rheostat controlled lights mounted in four of the test section "flats" and in the test section plenum chamber so as to shine through the slots. Continuous and flash ultra-violet light is provided by up to three mercury arc lamps with ultra-violet filters mounted at various locations (as desired) in the test section plenum chamber. Flash ultra-violet or flash white light is provided by up to three 2000 candle-second photographic power supplies with up to six lampheads (with or without ultra-violet filters) per supply which can be mounted at various locations in the test section plenum chamber.

References

VI-1. Berrier, B. L.; Leavitt, L. D.; and Bangert, L.S.: Operating Characteristics of the Multiple Critical Venturi System and Secondary

Calibration Nozzles Used for Weight-Flow Measurements in the Langley 16-Foot Transonic Tunnel. NASA TM 86405, 1985.

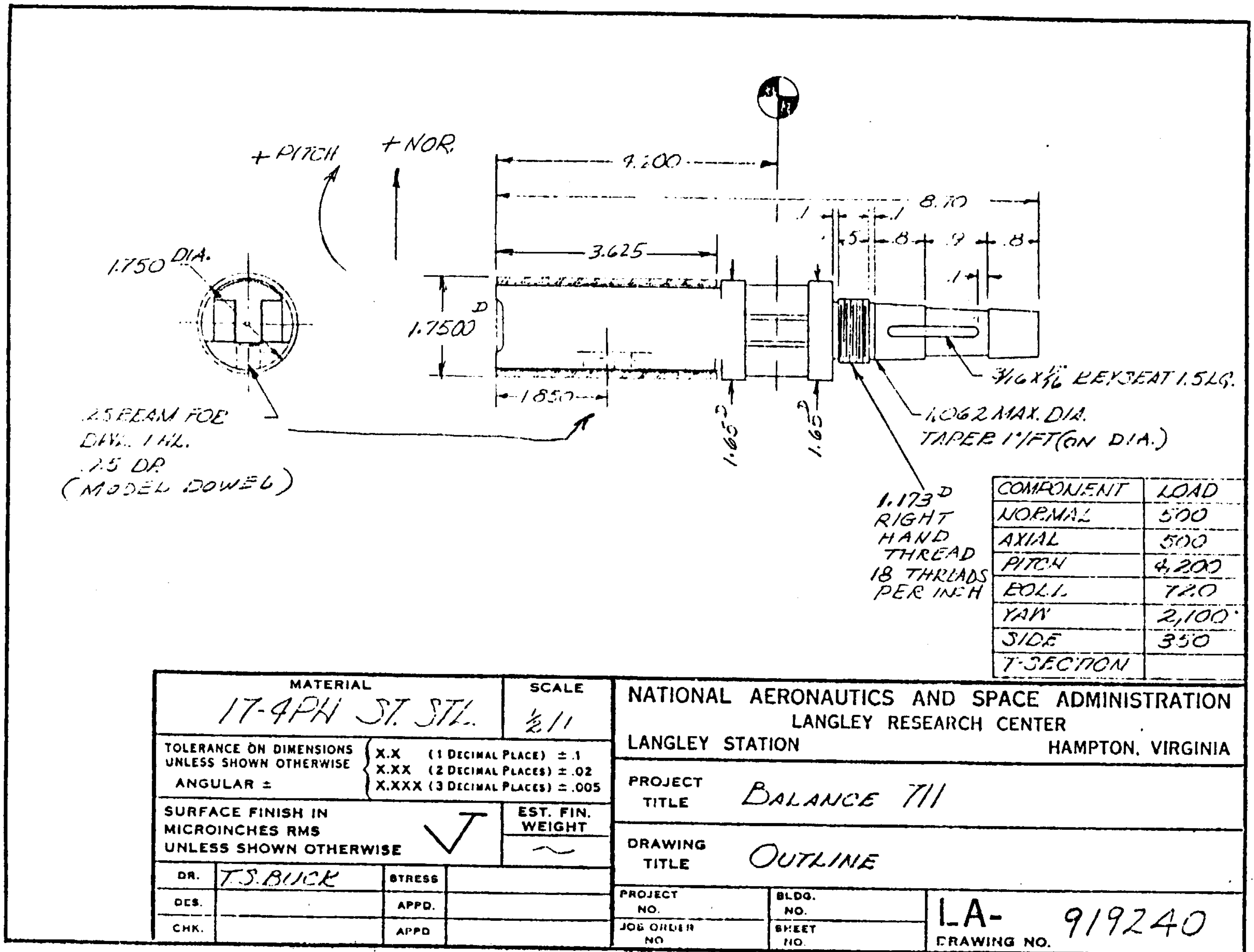


Figure VI-1. 711 Balance.

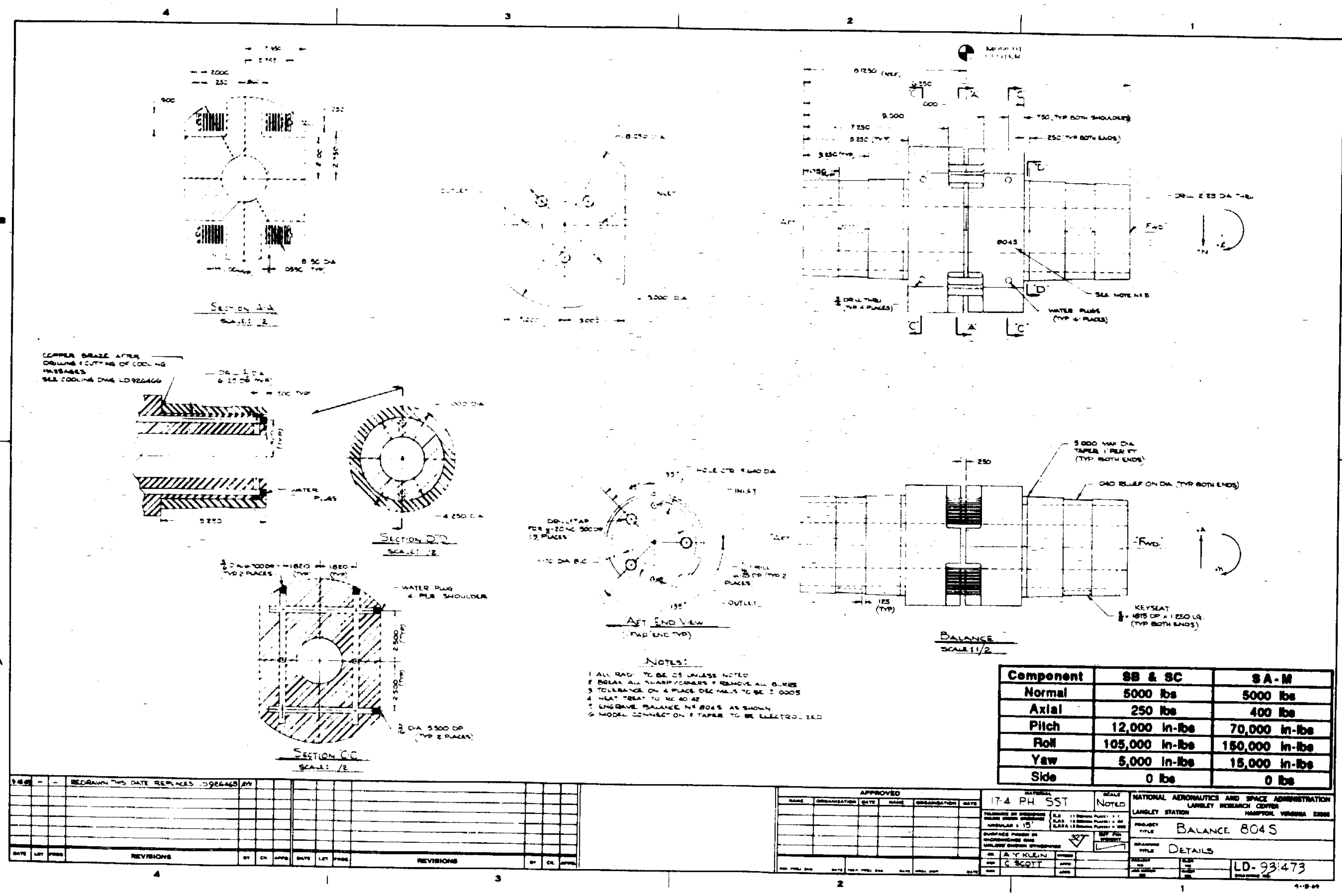
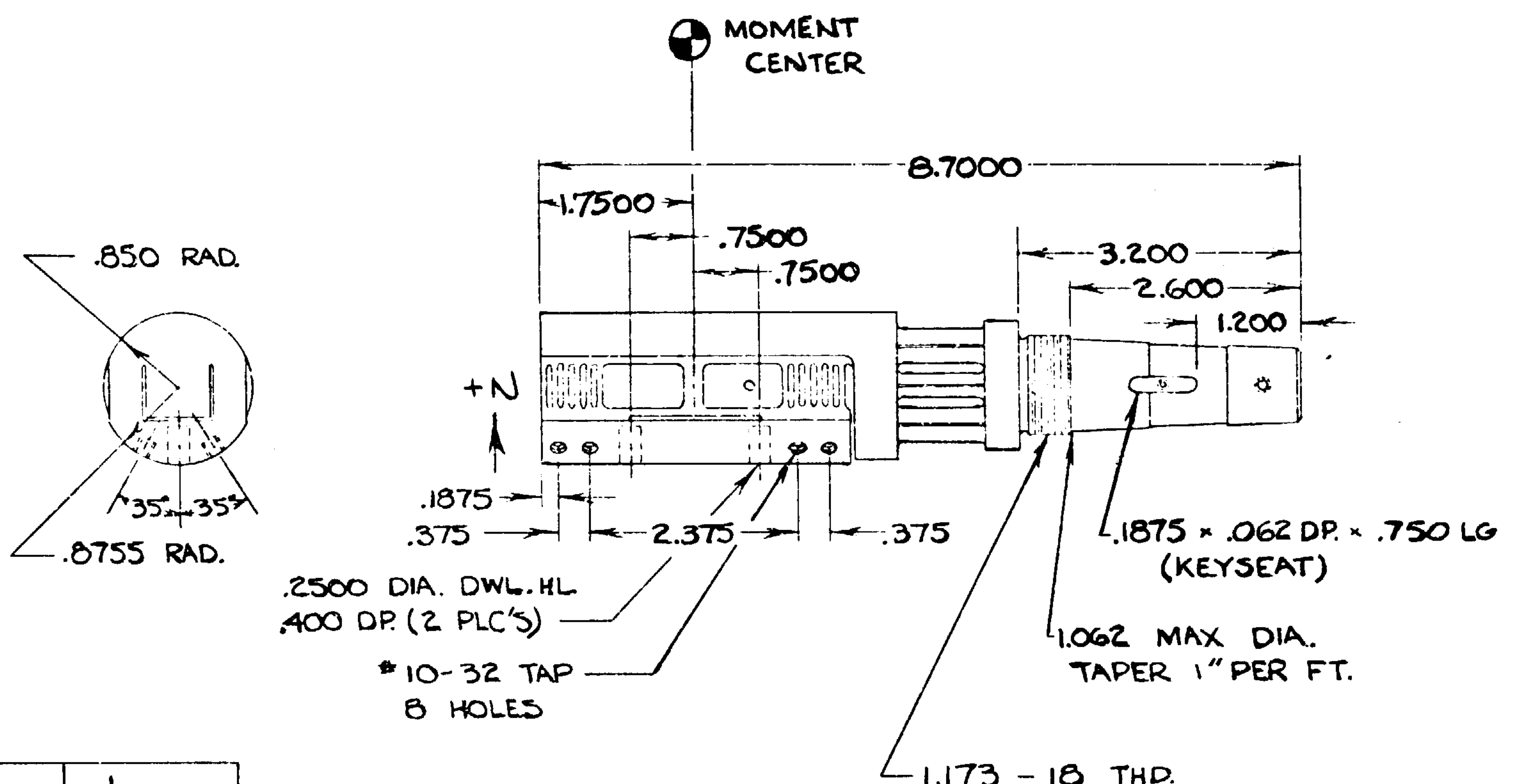


Figure VI-2. 804 Balance.



Component	Load
Normal	500#
Axial	800#
Pitch	2000"#

MATERIAL: MARAGING - 18% NICKLE	SCALE: 1/2	NATIONAL AERONAUTICS AND SPACE ADMINISTRATION LANGLEY RESEARCH CENTER LANGLEY STATION HAMPTON, VIRGINIA 23365	
TOLERANCE ON DIMENSIONS UNLESS SHOWN OTHERWISE ANGULAR ± ~	X.X (1 DECIMAL PLACE) ± .1 X.XX (2 DECIMAL PLACES) ± .02 X.XXX (3 DECIMAL PLACES) ± .005	PROJECT TITLE	BALANCE 1614
SURFACE FINISH IN MICROINCHES RMS UNLESS SHOWN OTHERWISE	EST. FIN. WEIGHT ~	DRAWING TITLE	OUTLINE
DR. T.W. BRAY	STRESS	PROJECT NO. / BLDG. NO.	LA-931309
DES. WOODDELL	APPD.	JOB ORDER NO / SHEET NO.	DRAWING NO.
CHK.	APPD.		

Figure VI-3. 1614 Balance.

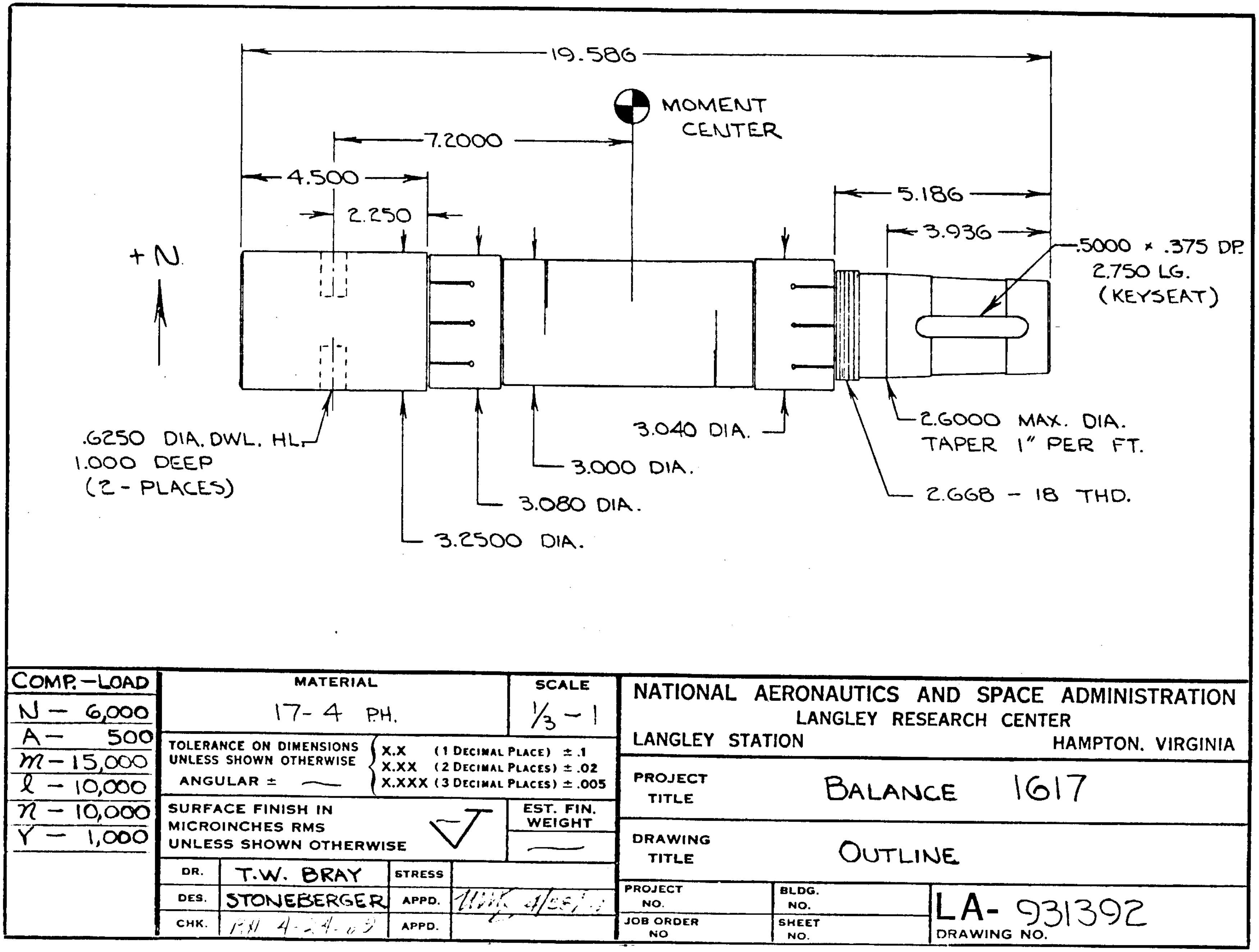


Figure VI-4. 1617 Balance.

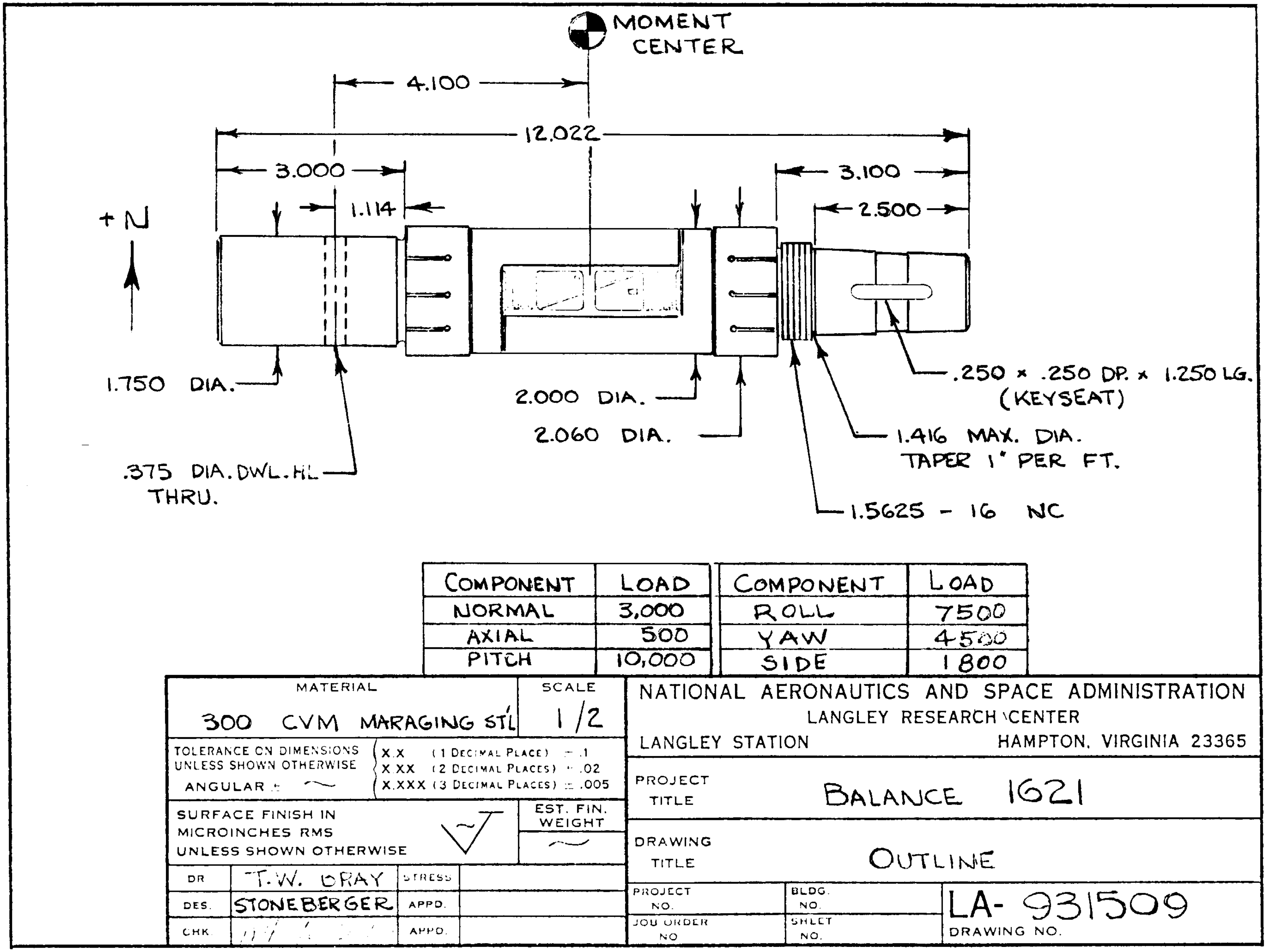


Figure VI-5. 1621 Balance.

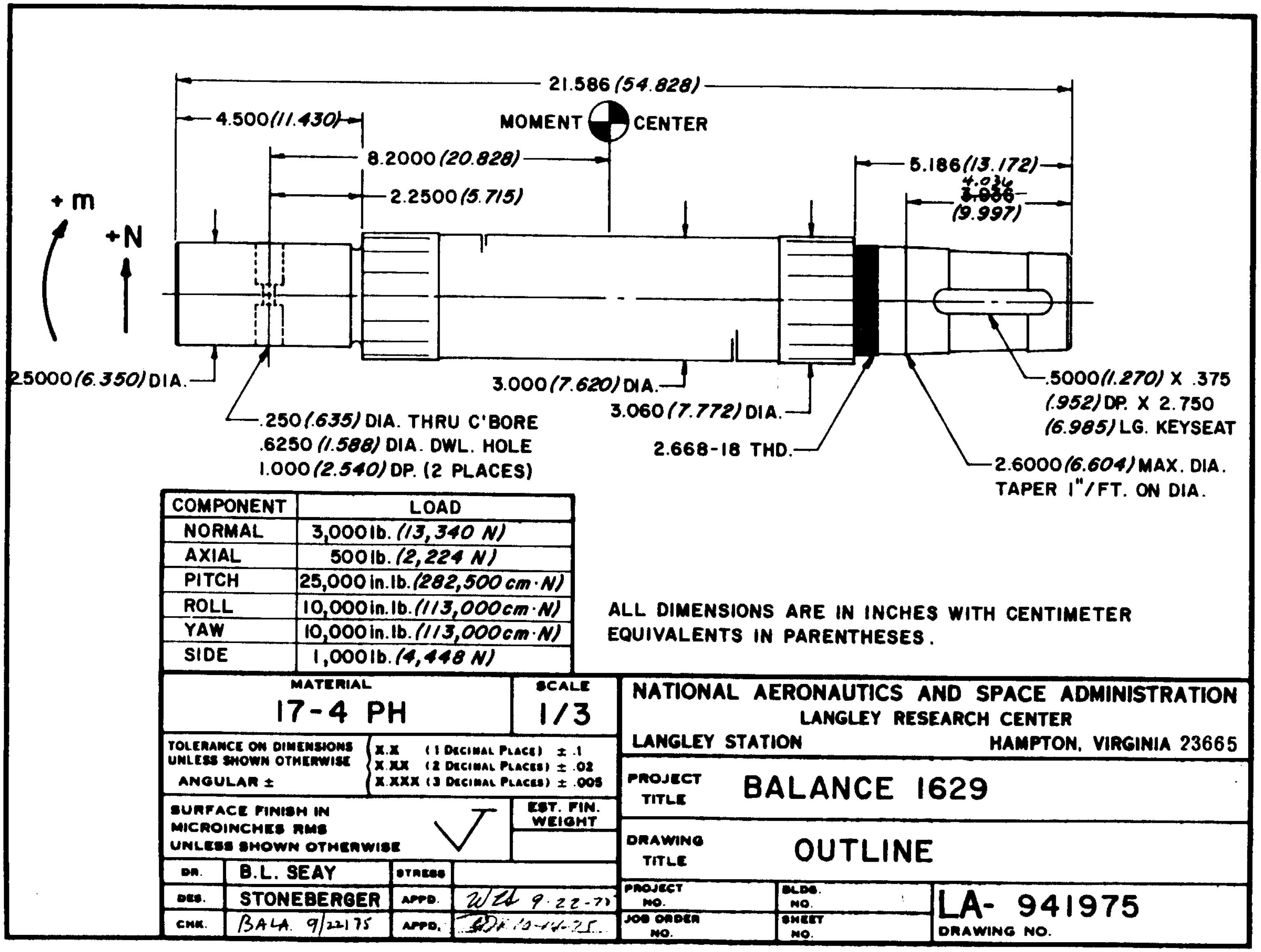
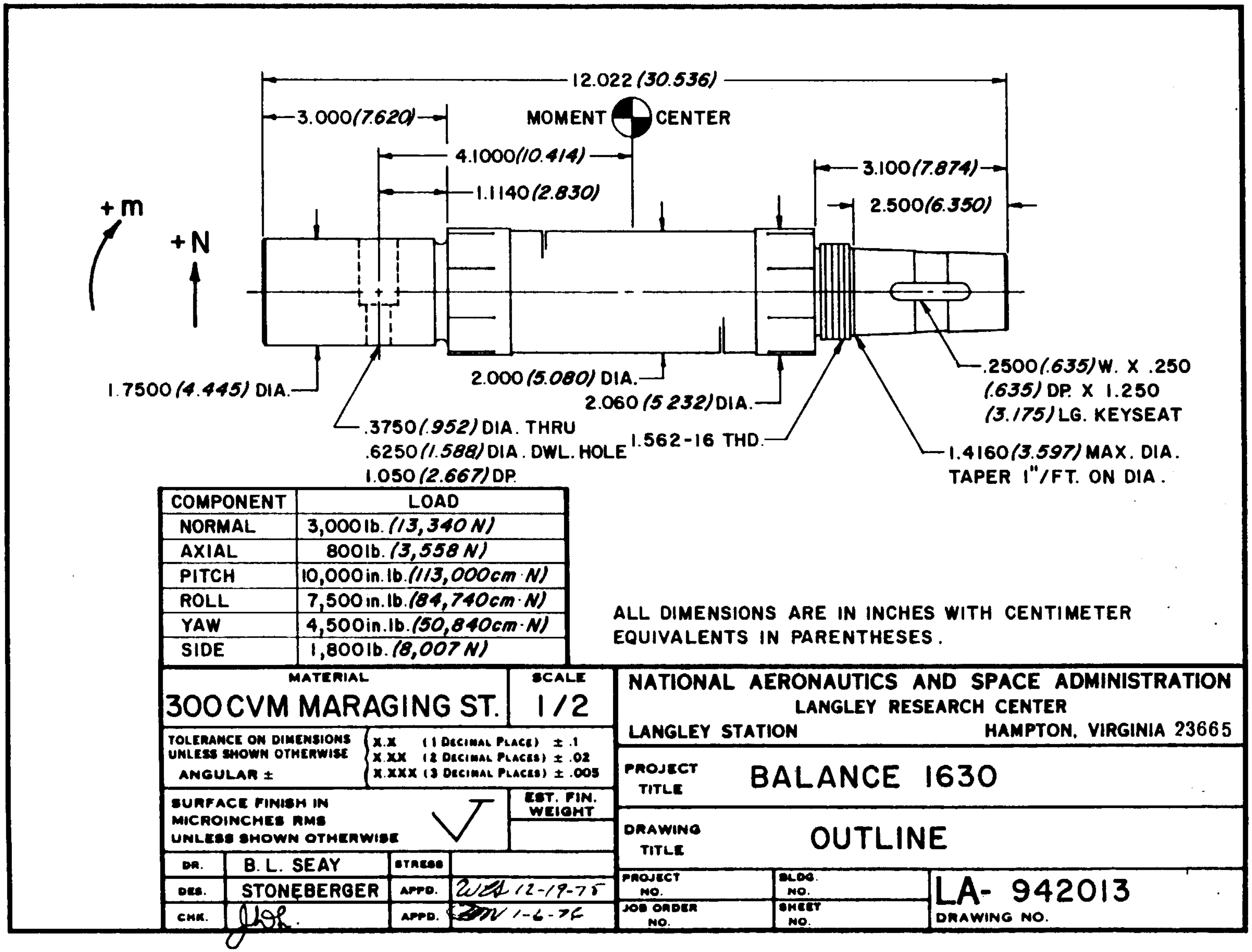
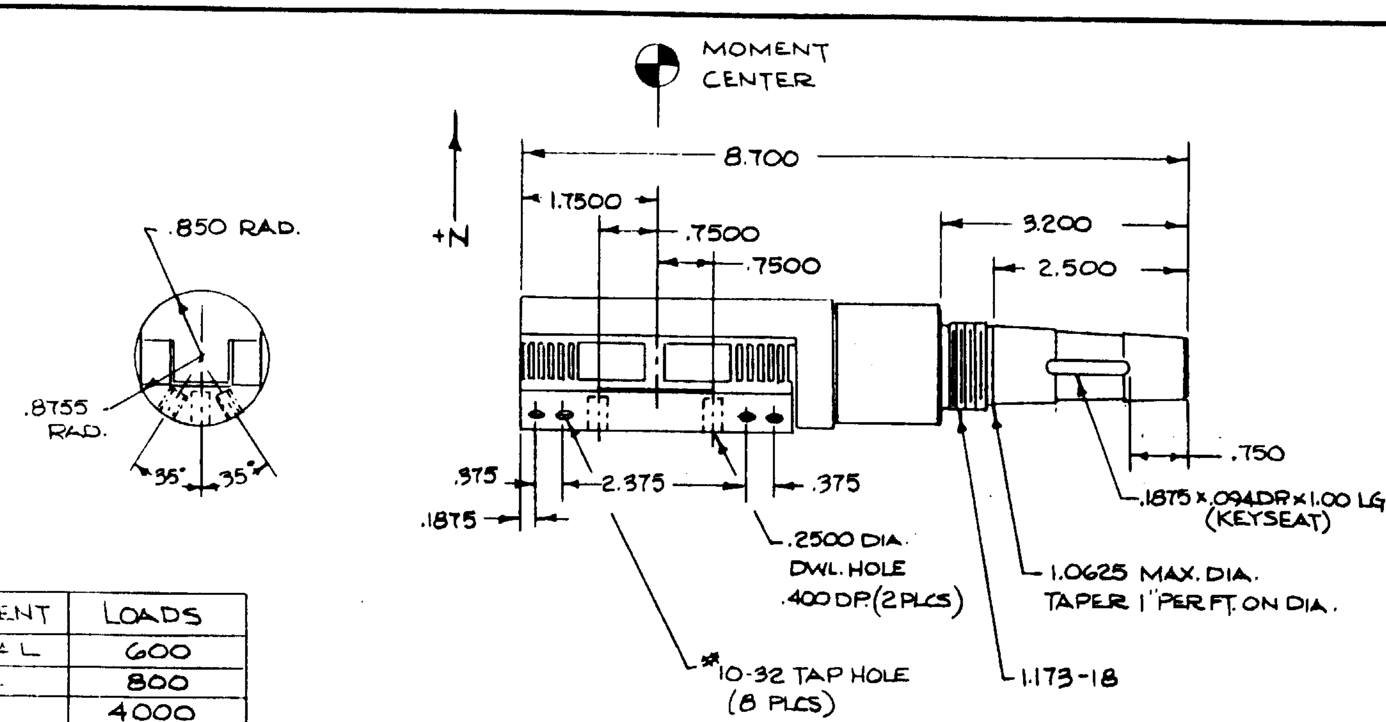


Figure VI-6. 1629 Balance.



COMPONENT	LOAD
NORMAL	3,000 lb. (13,340 N)
AXIAL	800 lb. (3,558 N)
PITCH	10,000 in. lb. (113,000 cm·N)
ROLL	7,500 in. lb. (84,740 cm·N)
YAW	4,500 in. lb. (50,840 cm·N)
SIDE	1,800 lb. (8,007 N)

Figure VI-7. 1630 Balance.



COMPONENT	LOADS
NORMAL	600
AXIAL	800
PITCH	4000
ROLL	1000
YAW	2000
SIDE	600

MATERIAL: 300 CVM MARAGING — SCALE: 1/2

TOLERANCE ON DIMENSIONS UNLESS SHOWN OTHERWISE: X.X (1 DECIMAL PLACE) ± .1; X.XX (2 DECIMAL PLACES) ± .02; X.XXX (3 DECIMAL PLACES) ± .005

ANGULAR ±

SURFACE FINISH IN MICROINCHES RMS UNLESS SHOWN OTHERWISE — EST. FIN. WEIGHT

DR. A.Y. KLEIN — STRESS

DES. STONEBERGER — APPD. WEL 9-16-69

CHK. RH 9-9-69 — APPD.

NATIONAL AERONAUTICS AND SPACE ADMINISTRATION
LANGLEY RESEARCH CENTER
LANGLEY STATION — HAMPTON, VIRGINIA 23365

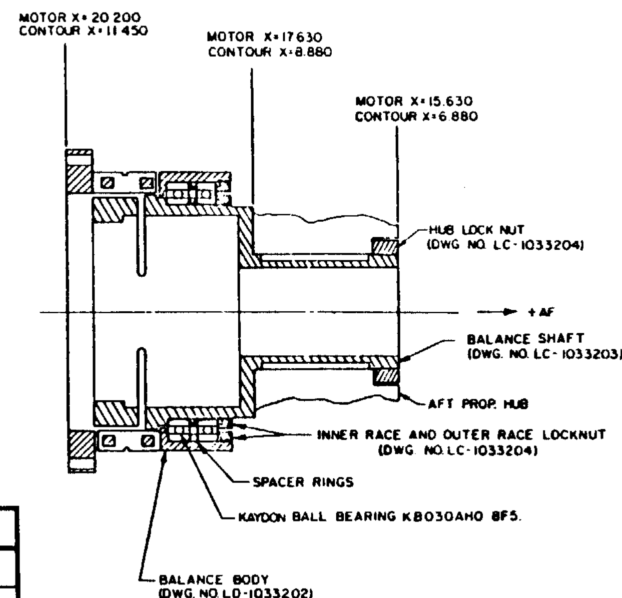
PROJECT TITLE: BALANCE 1631

DRAWING TITLE: OUTLINE (WITH SHIELD)

PROJECT NO. — BLDG. NO. — JOB ORDER NO. — SHEET NO.

LA-931484
DRAWING NO.

Figure VI-8. 1631 Balance.



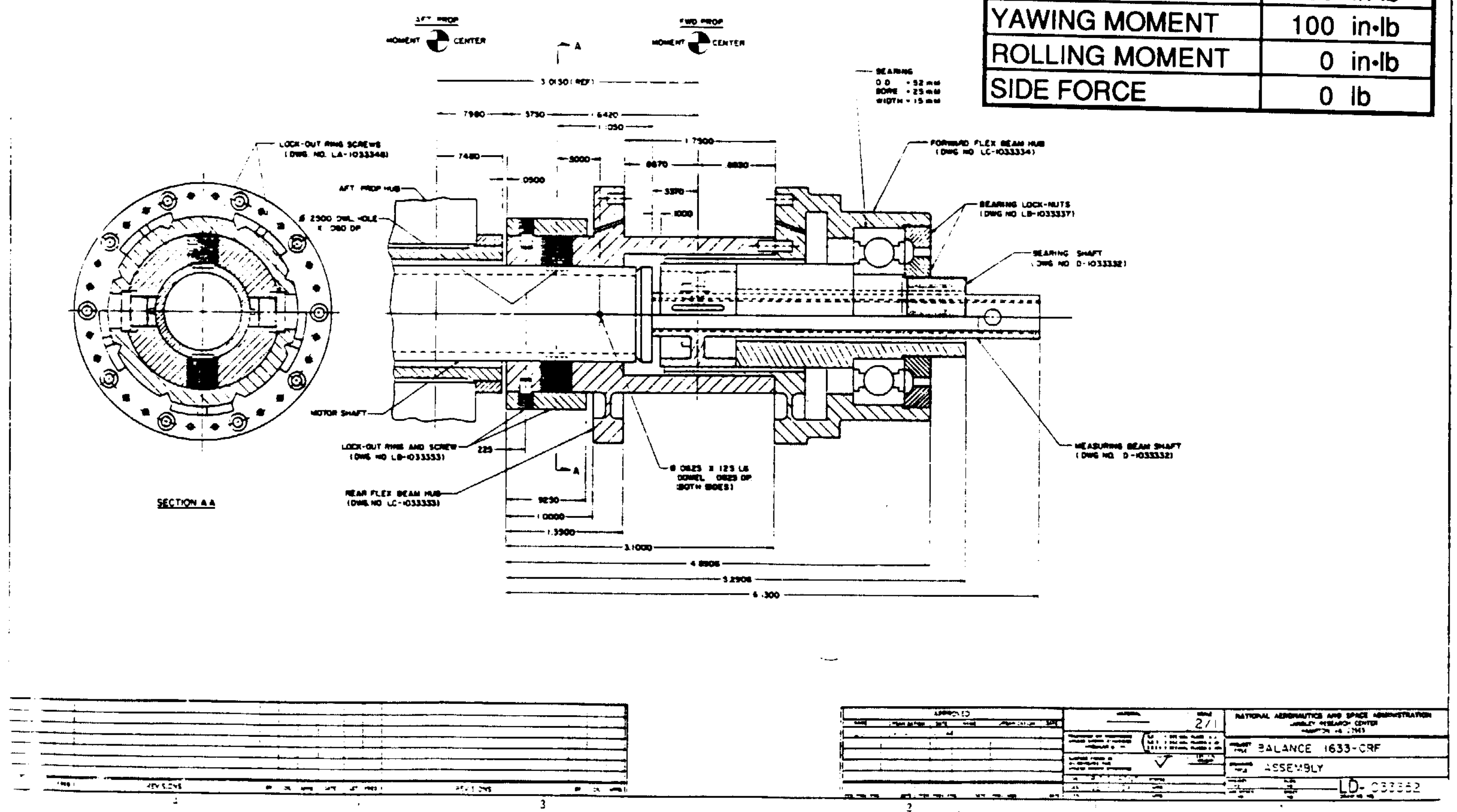
COMPONENT	LOAD
NORMAL FORCE	100 lb
AXIAL FORCE	150 lb
PITCHING MOMENT	200 in•lb
YAWING MOMENT	100 in•lb
ROLLING MOMENT	0 in•lb
SIDE FORCE	0 lb

REVISIONS
APPROVED
MATERIAL
SCALE 1/1
NATIONAL AERONAUTICS AND SPACE ADMINISTRATION
LANGLEY RESEARCH CENTER
HAMPTON, VIRGINIA 23665
PROJECT TITLE: BALANCE 1633-CRA
DRAWING TITLE: BACK BALANCE ASSEMBLY (1633-CRA)
C K SERVAIS
N PATEL
LC- 1033I9-9

(a) CRA

Figure VI-9. 1633 Balance

COMPONENT	LOAD
NORMAL FORCE	100 lb
AXIAL FORCE	150 lb
PITCHING MOMENT	200 in•lb
YAWING MOMENT	100 in•lb
ROLLING MOMENT	0 in•lb
SIDE FORCE	0 lb



(b) CRF

Figure VI-9. Concluded.

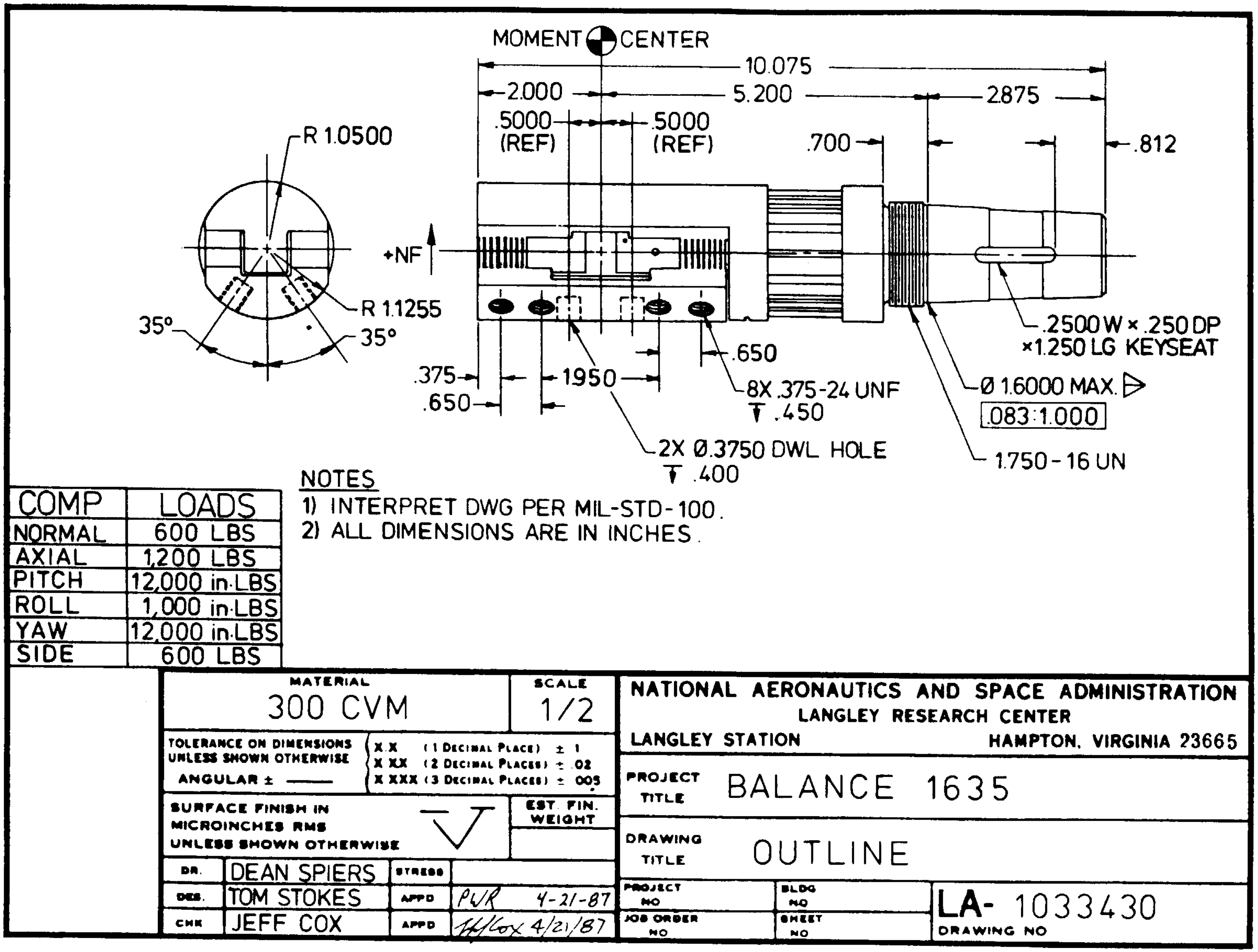


Figure VI-10. 1635 balance system. (All dimensions are in inches.)

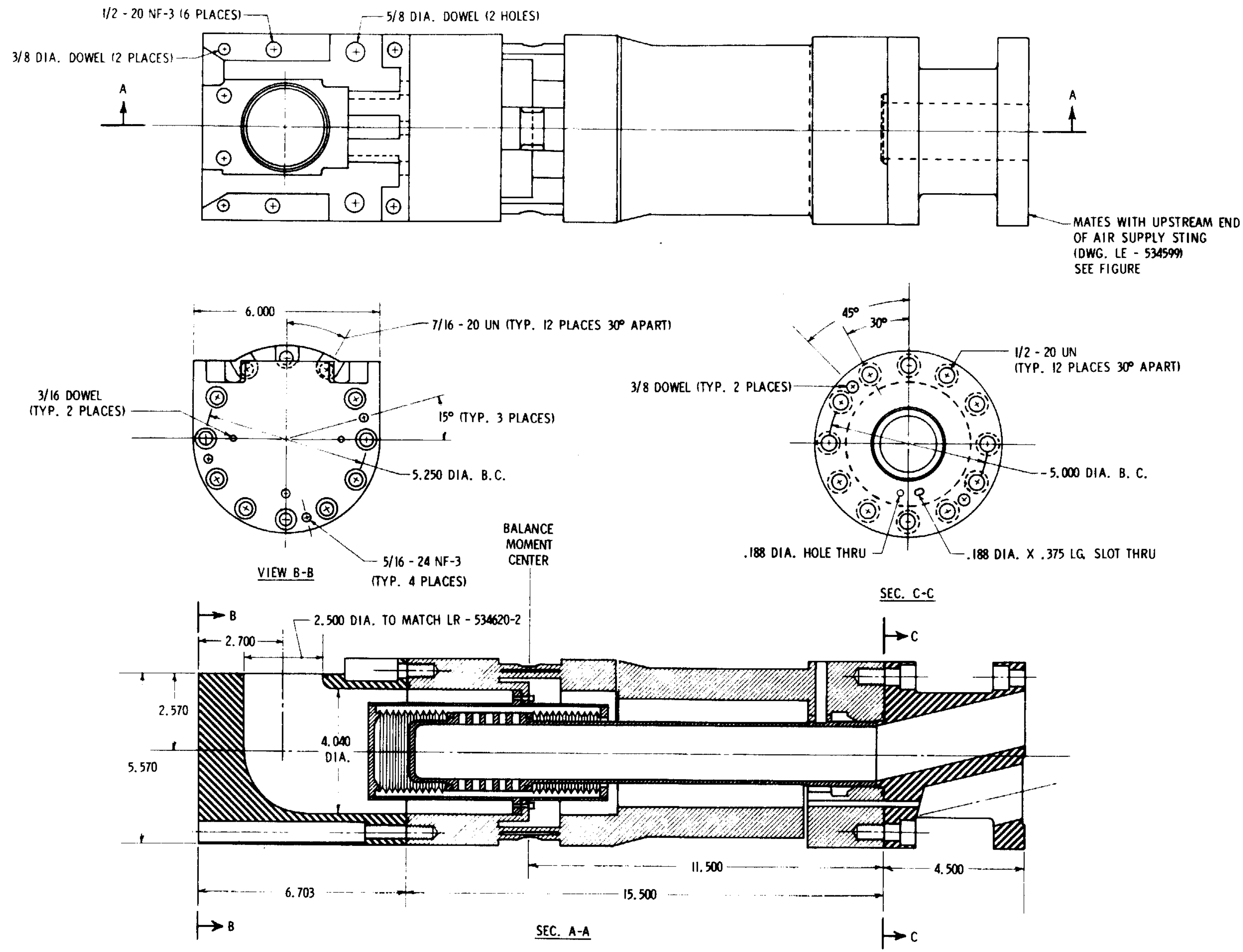


Figure VI-11. 1627 balance system. (All dimensions are in inches.)

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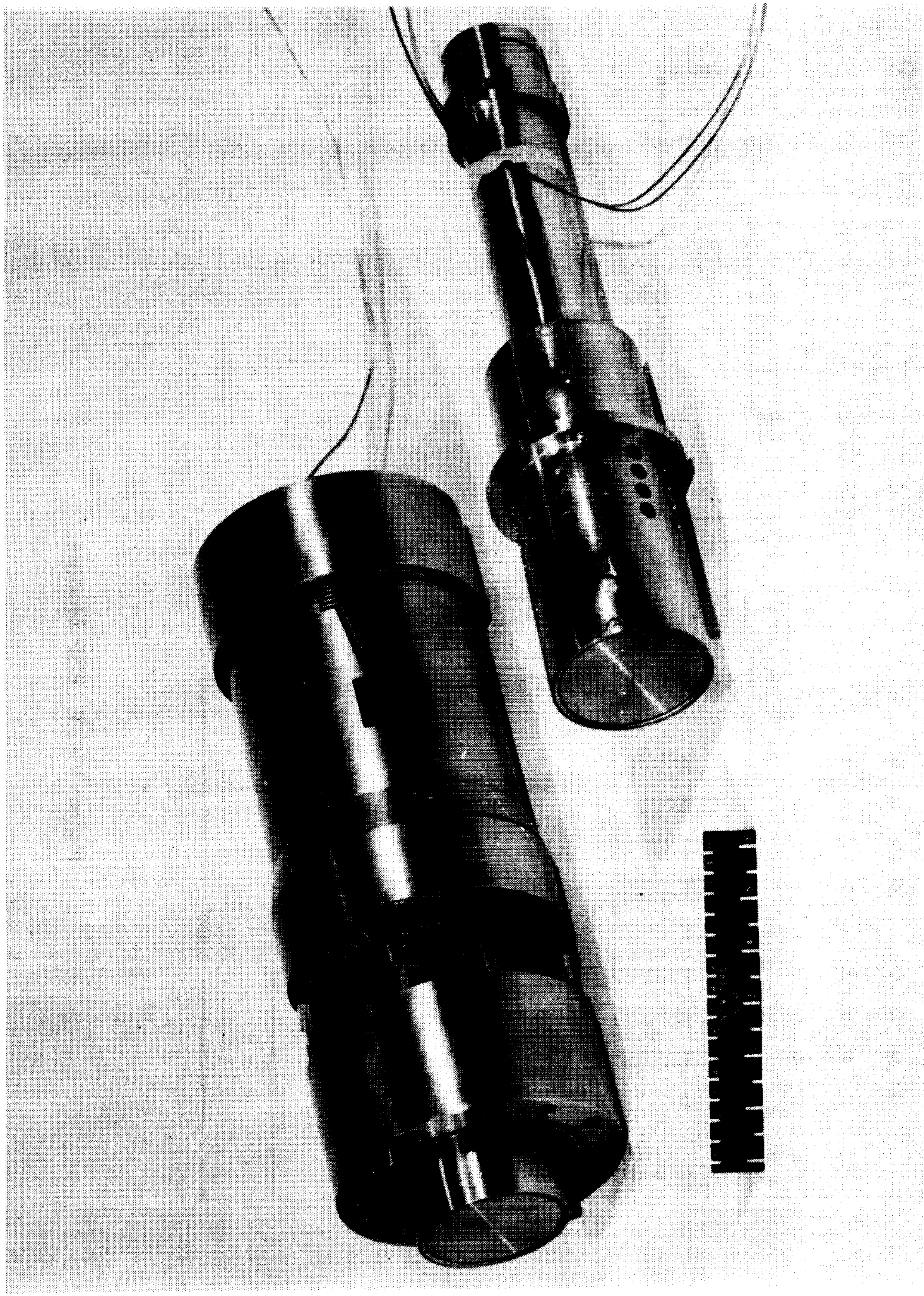


Figure VI-12. 1627 balance and extra air supply tube/bellows assembly.

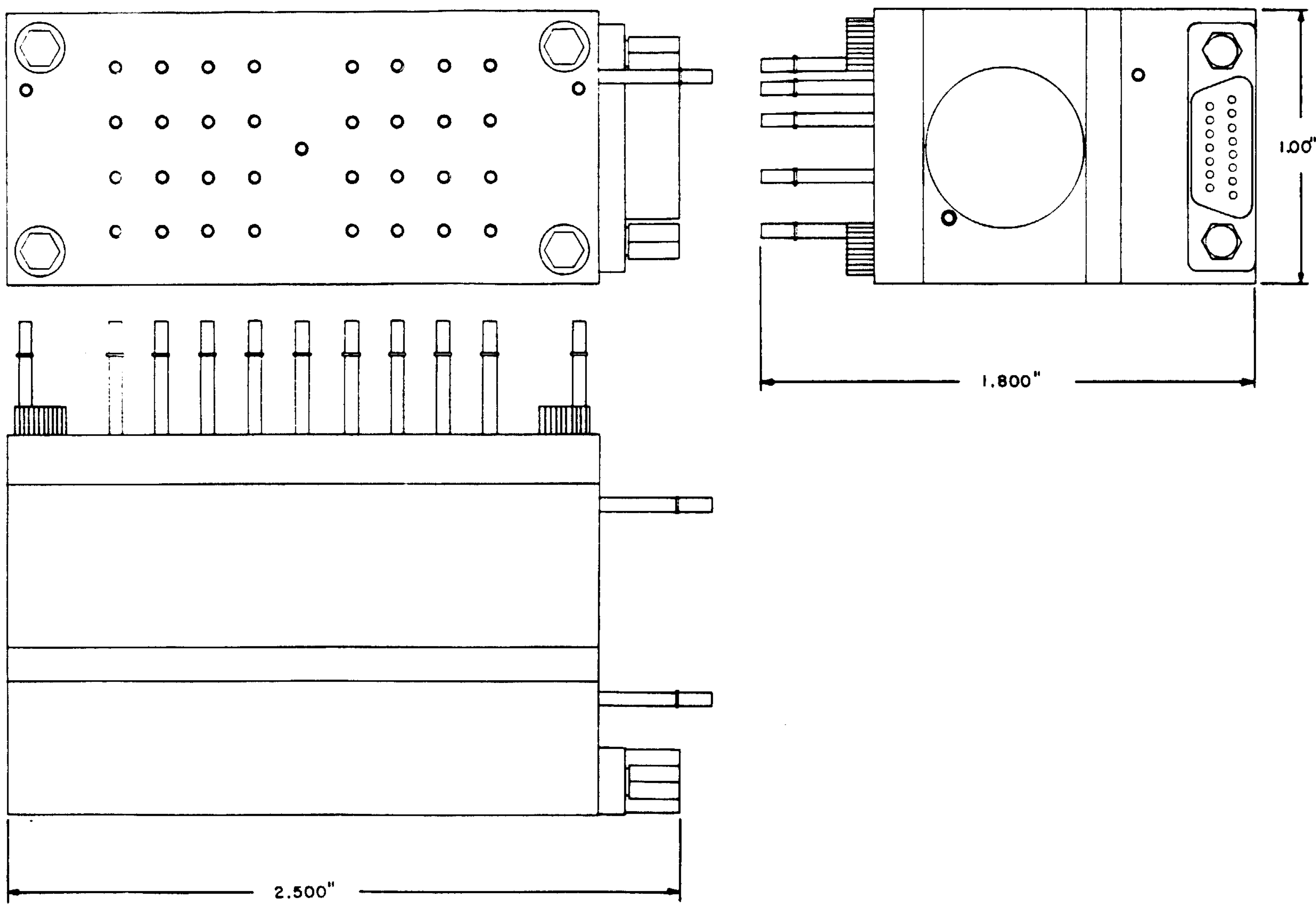
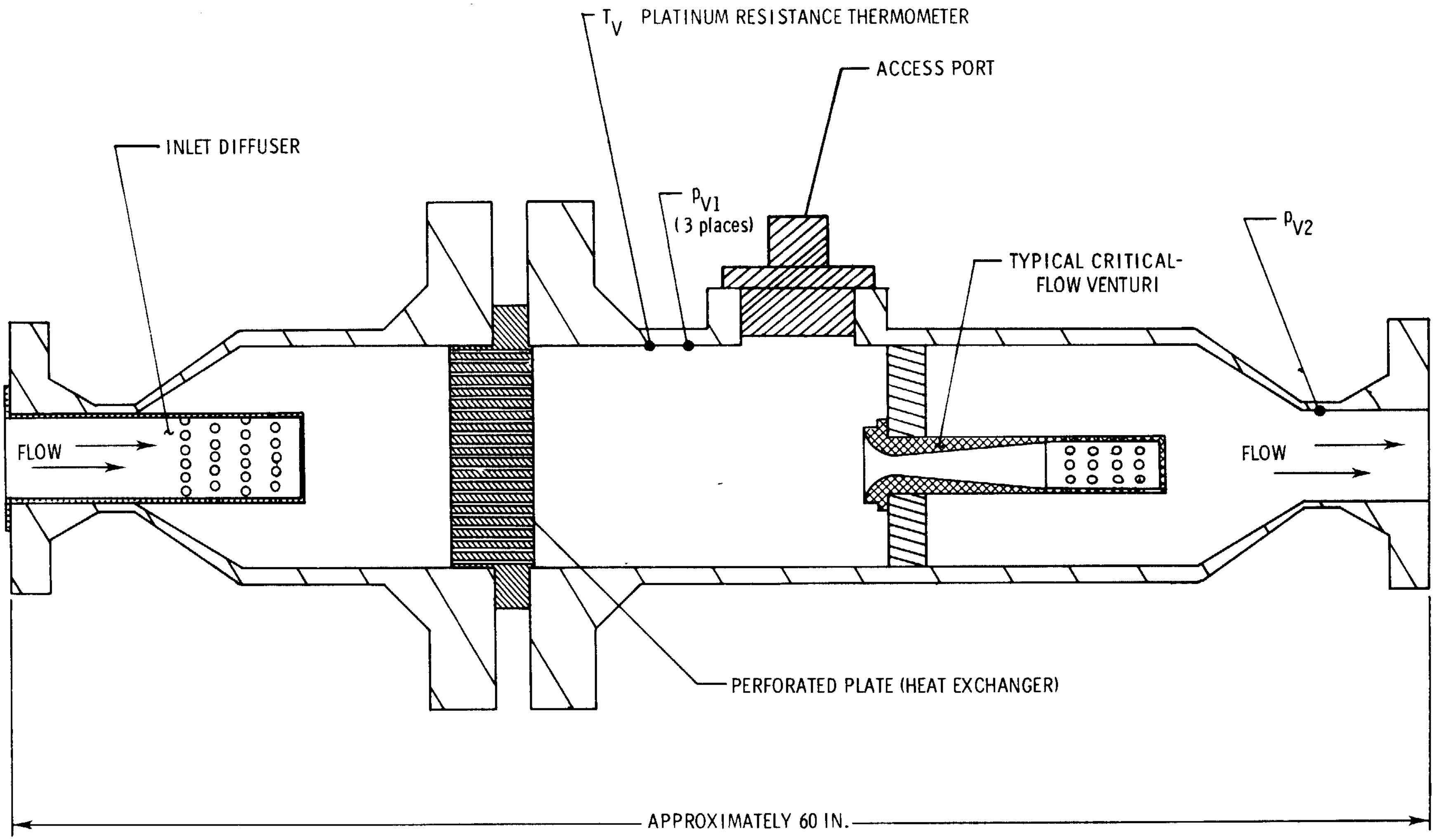


Figure VI-13. ESP Module.



(a) Sketch of multiple critical venturi system.

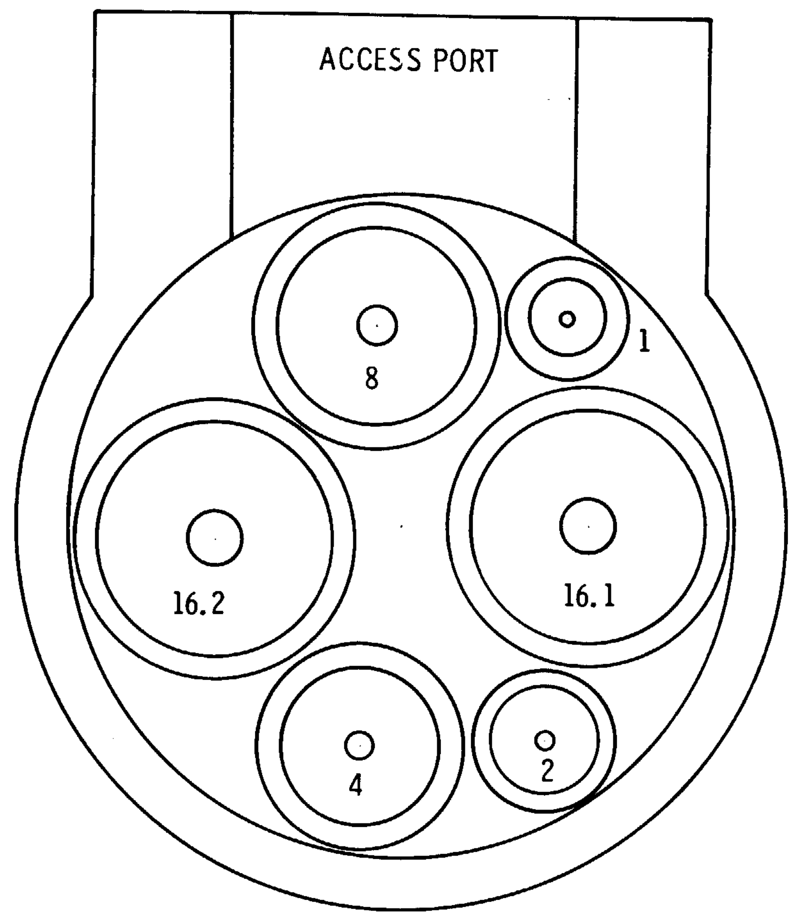
Figure VI-14. Geometry of multiple critical venturi system.

L-85-110

(b) Multiple critical venturi system.

Figure VI-14. Continued.

VENTURI GEOMETRY		
VENTURI NO.	THROAT DIA., IN.	THROAT AREA, IN^2
1	0.1877	0.0277
2	0.2639	0.0547
4	0.3741	0.1099
8	0.5281	0.2190
16.1	0.7475	0.4388
16.2	0.7478	0.4392



(c) Individual venturi geometry and orientation.

Figure VI-14. Concluded.

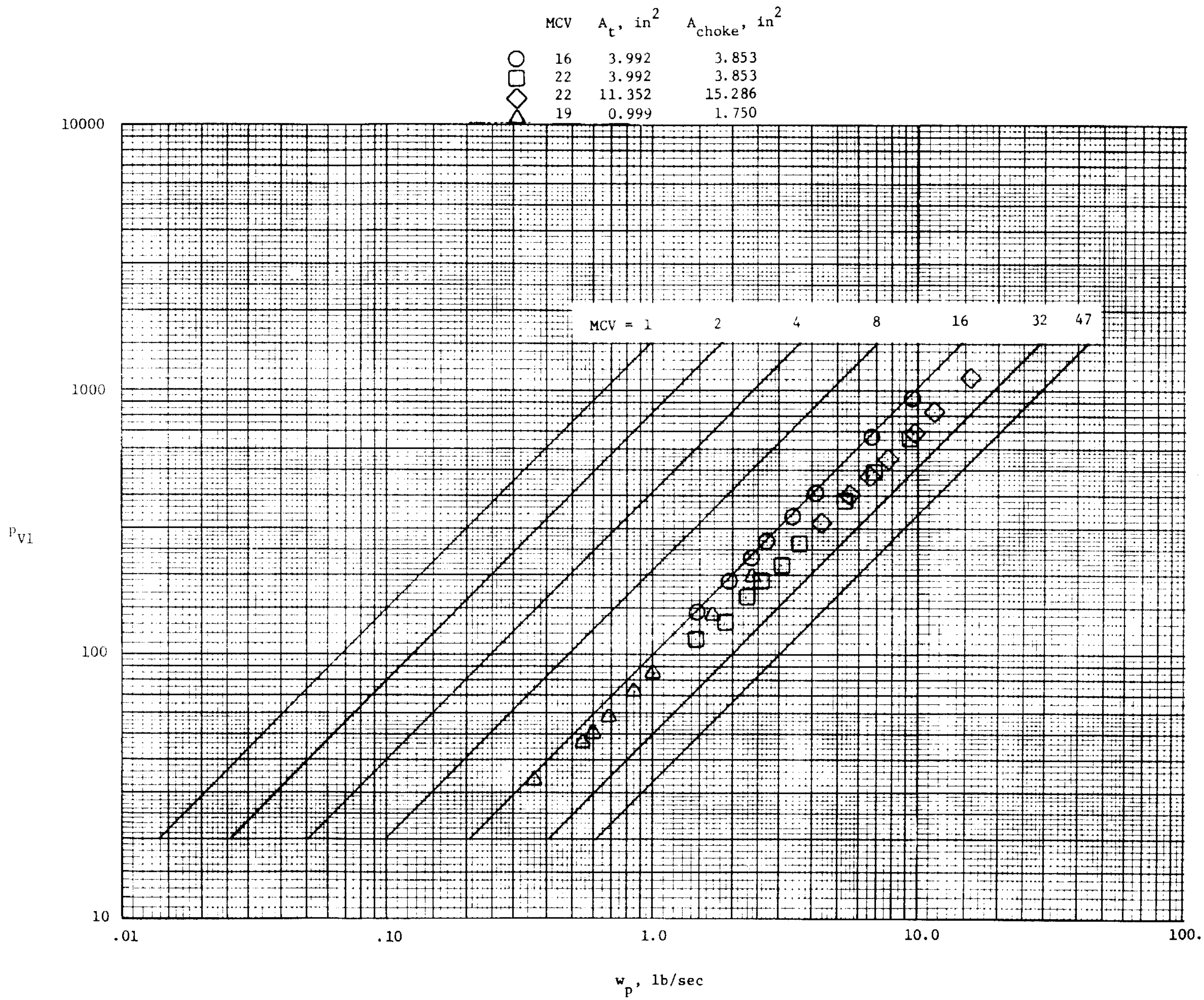


Figure VI-15. Multiple critical venturi operating envelope.

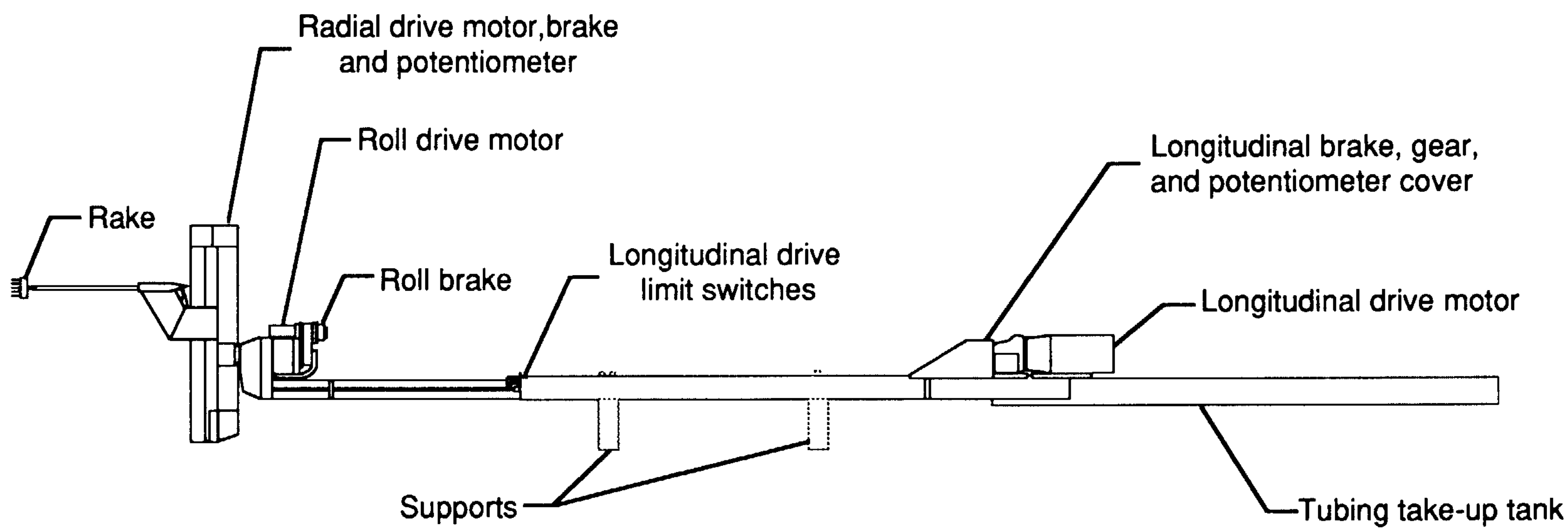


Figure VI-16. Remotely controlled translating survey mechanism.

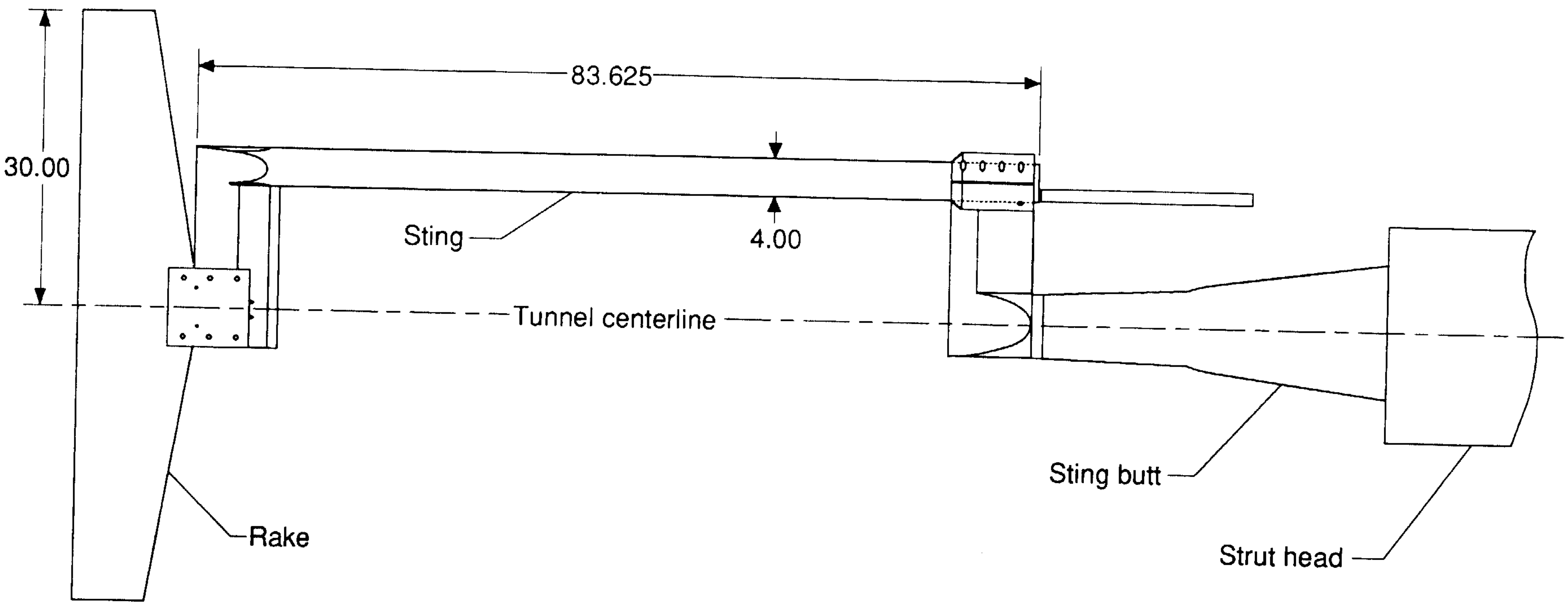


Figure VI-17. Translating sting attached to strut head. Dimensions shown in inches.

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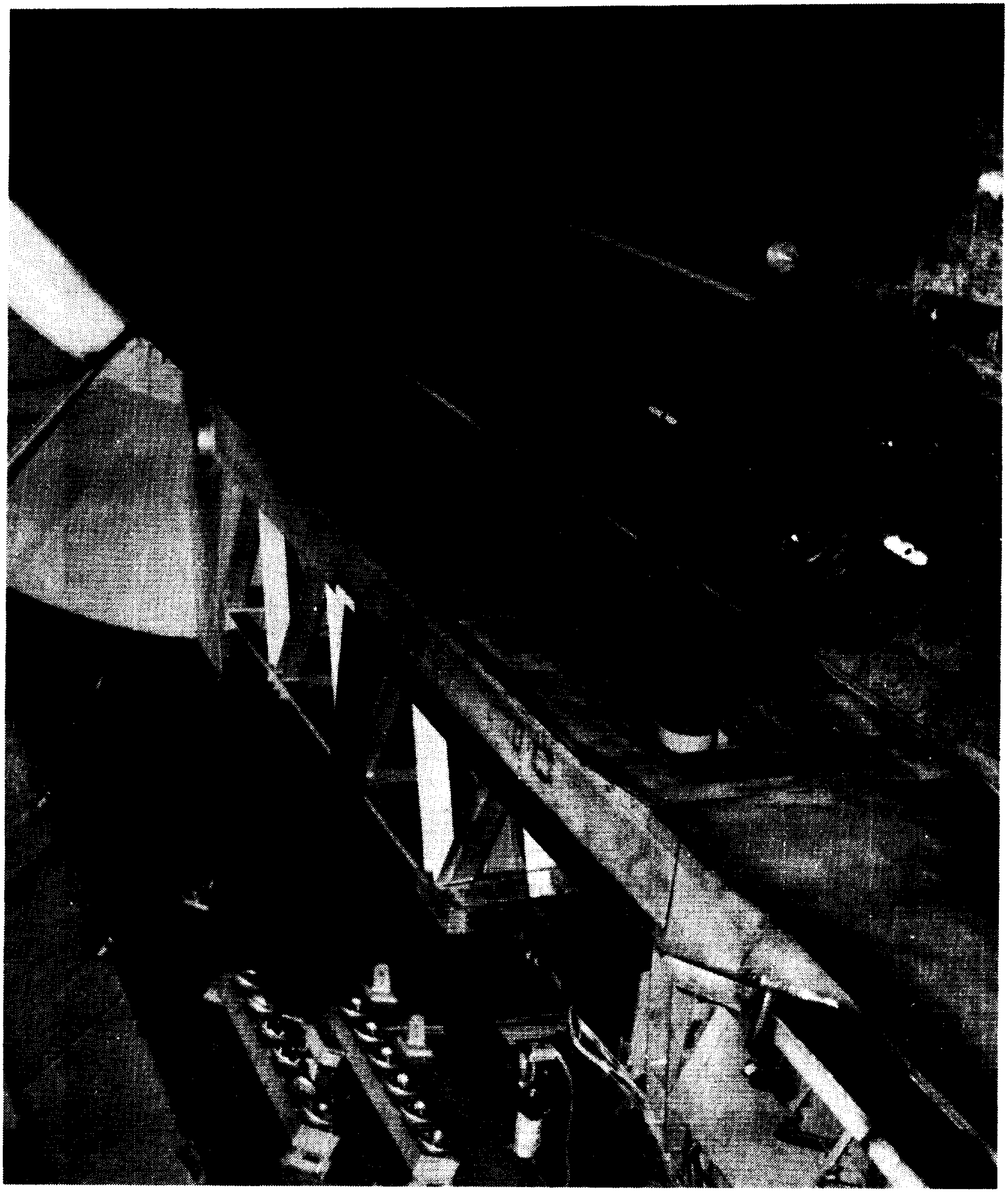

Figure VI-18. LDV scanning mechanism installed in the 16 T.T. test section plenum.

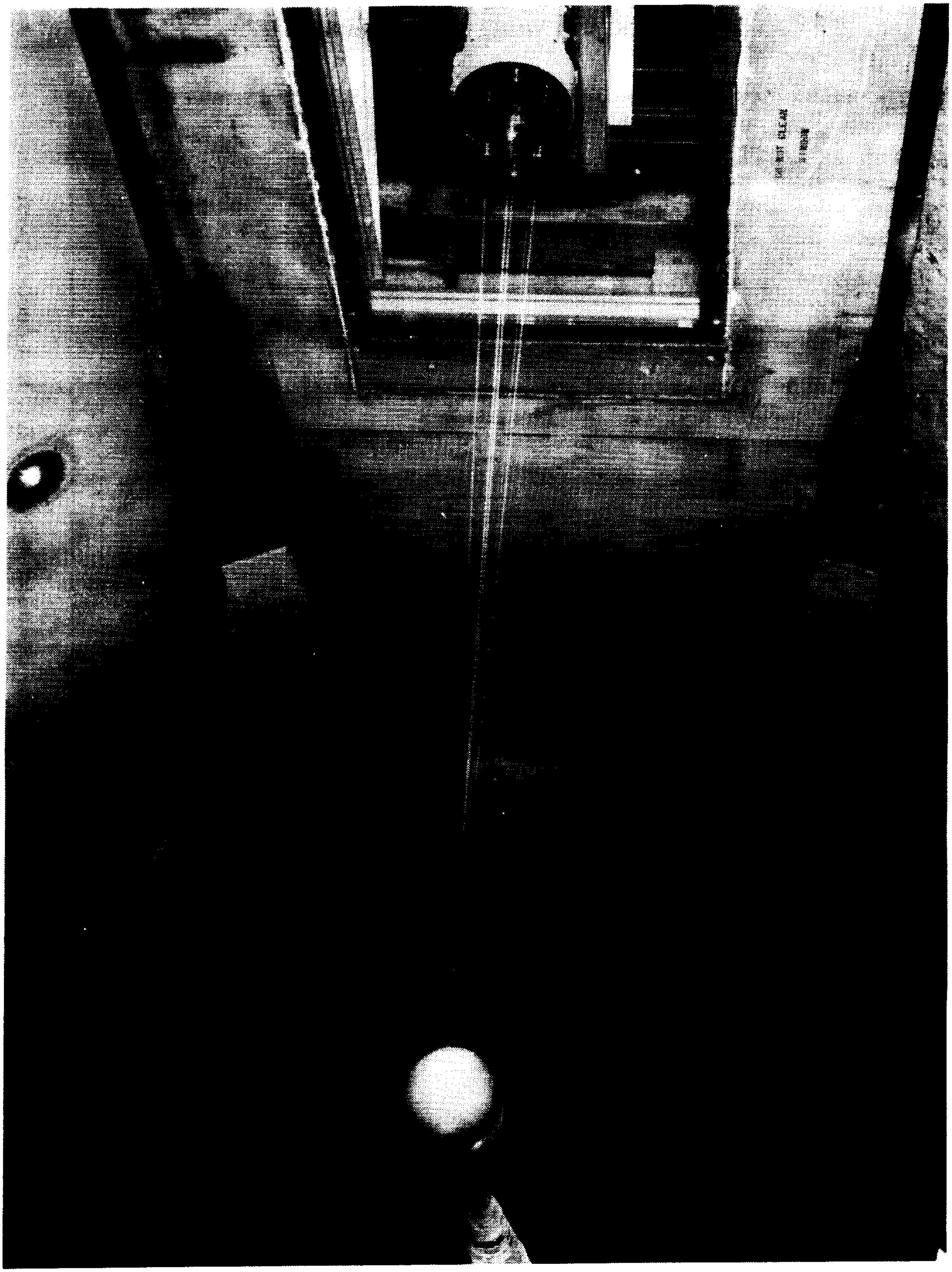

Figure VI-19. LDV system in operation.

SECTION VII - 16-Foot Tunnel Computer Facilities

The 16-Foot Transonic Tunnel computer complex contains all necessary equipment required to acquire, reduce, and analyze wind tunnel test data. Both the data acquisition and data reduction systems will be discussed briefly herein, however, both hardware and software are in a continuous state of improvement. It it therefore imperative that data reduction and analysis requirements be discussed with personnel of the Propulsion Aerodynamics Branch as early as possible in the test planning cycle.

A. Data Acquisition System.- The central purpose of the wind tunnel facility is to acquire data on the forces, pressures, and temperatures acting on models in simulated flight. The 16-Foot Transonic Tunnel data acquisition system is comprised of four high speed measuring and recording devices:

1	Modcomp 3287/Neff - Tunnel DAU
2	Modcomp 3287 - Tunnel automation
3	Modcomp Classic - Model Prep. DAU
4	Modcomp Classic - Ground Stand DAU

These systems have been designed to meet the special needs of the 16-Foot Transonic Tunnel Complex (see figure VII-1).

1). Modcomp/Neff.- Test control, data acquisition, and real time data reduction for the test operational mode of the tunnel are performed utilizing a Modcomp 3287 computer complex interfaced with a Neff analog amplifier conditioning unit. The system can accommodate the following inputs:

128 analog (measured in microvolts) pressure, force and
temperature
15 digital (measured in counts)
24 multiple pressure scanners (scanivalves)
512 ESP channels (16 ESP modules)

These inputs can be recorded with a sampling rate up to 80 samples per second. The normal rate is 10 samples per second. Data can be acquired in several modes of real time; i.e.,

CRT display with hard copy output
Digital panel display
Line printer (hard copy)
Magnetic tape (raw data or SIF format)

Input information required from the user prior to tunnel entry includes:

Special equations (functions not included in ref. VII-1)
Areas (including inlet and nozzle areas, integration areas, wetted areas, etc.)
Instrumentation requirements
Data printout requirements
Data transmittal requirements

Real time data output includes engineering units of all test measurements, computed aerodynamic force and moment coefficient, and propulsive parameters. Finalized data reduction (batch mode) is generally available upon completion of a test run (however, the data reduction processing is usually accomplished on separate computer hardware which will be discussed later). This mode is the same as real time with the exception that both initial and final data zeros are used to reduce raw data input into final forces, moments, and coefficients. Approximately 2200 bits of information per data point can be output by this mode.

2). Modcomp.- Tunnel system automation for the of the tunnel are performed utilizing a Modcomp 3287 computer. Tunnel conditions, model attitude and model propulsive setting are controlled through this system with micro-proccessor interfaces.

3). Modcomp Classic.- Test control, data acquisition, and real time data reduction for the aerodynamic-propulsive-airframe integration test operational mode of the model preparation room are performed utilizing a Modcomp Classic computer interfaced with a Neff analog amplifier conditioning unit. The system can accommodate the same inputs as the tunnel computer.

4). Modcomp Classic.- Test control, data acquisition, and real time data reduction for the aerodynamic-propulsive-airframe integration test operational mode of the Static Test Facility in building 1234 are performed utilizing a Modcomp Classic computer complex interfaced with a Neff analog amplifier conditioning unit. The system can accommodate the same inputs as the tunnel computer system.

B. Data Reduction System.- After wind tunnel raw data has been acquired, final data reduction, analysis, and postprocessing functions are accomplished either on the branch local VAX network or at the LaRC Central Computing site. The local VAX network is located in building 1146 and is capable of handling raw data from the Modcomp data acquisition computers. It consists of the following hardware:

1	MicroVAX II
2	VAXstation 2000
3	VAXstation 3

Several different network links exists between the data acquisition and data reduction (VAX, Modcomp and Central Computing) systems. The Modcomp and VAX communicate via a medium speed hyper-channel link for the transfer of raw data files and tunnel set-up decks between the systems. The VAX and the Central site systems communicate primarily via a fiber-optic based ethernet link. The Central site and the VAX are also accessible via a low speed dial-up terminal server incorporated into Langley's voice computerized branch exchange. The three VAX systems running (at present) VMS V5.3 form an independent Local Area VAXcluster via an ethernet sub-loop for the sharing of disk space, software packages, and batch queues. Description of the Local Area VAXcluster follows:

1). MicroVAX II.- The primary function of the Digital Equipment Corporation (DEC) MicroVAX II computer (see figure VII-1 for a block diagram of the system) is the storage, reduction and analysis of wind tunnel data. Software programs are available for data reduction (reduce the raw data from the Modcomp into both engineering units and computed quantities) and analysis. The Data Analysis Subsystem, or DAS, software package contains extensive data handling features, plotting and printing capabilities, and other utilities which facilitate the complete data analysis process from reduced engineering units to final report figures. In addition, it is possible for a user to write software packages to provide further data reduction beyond the capabilities of the standard data reduction package that has been developed for the 16-Foot Transonic Tunnel. To exercise this option, the user must supply the software required.

2). VAXstation 2000.- This monochrome graphics workstation is a multi-windowing graphics environment permitting editing of data reduction files and pre-viewing plots created by the data reduction software.

3). VAXstation 3.- The primary function of the Digital Equipment Corporation VAXstation 3 computer (see figure VII-1 for a block diagram of the system) is the acquisition and reduction of laser velocimeter data. The computer system also provides real time control of the positioning equipment for the laser velocimeter. The laser velocimeter electronics and buffer interface convert the analog output from the laser velocimeter to a digital signal which is reduced to velocity components of the local flow by data reduction software resident in the computer. When this computer system is not being used for laser velocimeter experiments, it is available for use as an analysis tool for data obtained on the Neff/Modcomp computer the same as the MicroVAX II system above.

Duplicate data reduction and analysis software packages reside both on the VAX cluster and in the Langley Central Computers Complex to allow for the most flexible system possible. The LaRC Central Computer Complex has available for wind-tunnel data reduction a series of CYBER-860 mainframe computers running NOS 2.6 and CONVEX computers running UNICOS. A Hetra remote batch terminal is connected to the Central Complex for medium speed 132-column hard copy from any system at the Central Site or from the 16-Foot Tunnel control room.

Output data from the data reduction system may be supplied to the user in the following forms:

Line printer tab

Magnetic tape (user furnishes tape)

Optical disk (user furnishes disk)

Plots (special handling)

References

VII-1. Mercer, Charles E.; Berrier, Bobby L.; Capone, Francis J.; Grayston, Alan M.; and Sherman, C. D.: Computations for the 16-Foot Transonic Tunnel-NASA, Langley Research Center, Revision 1. NASA TM-86319, 1987. (Supersedes NASA TM-86319, 1984)

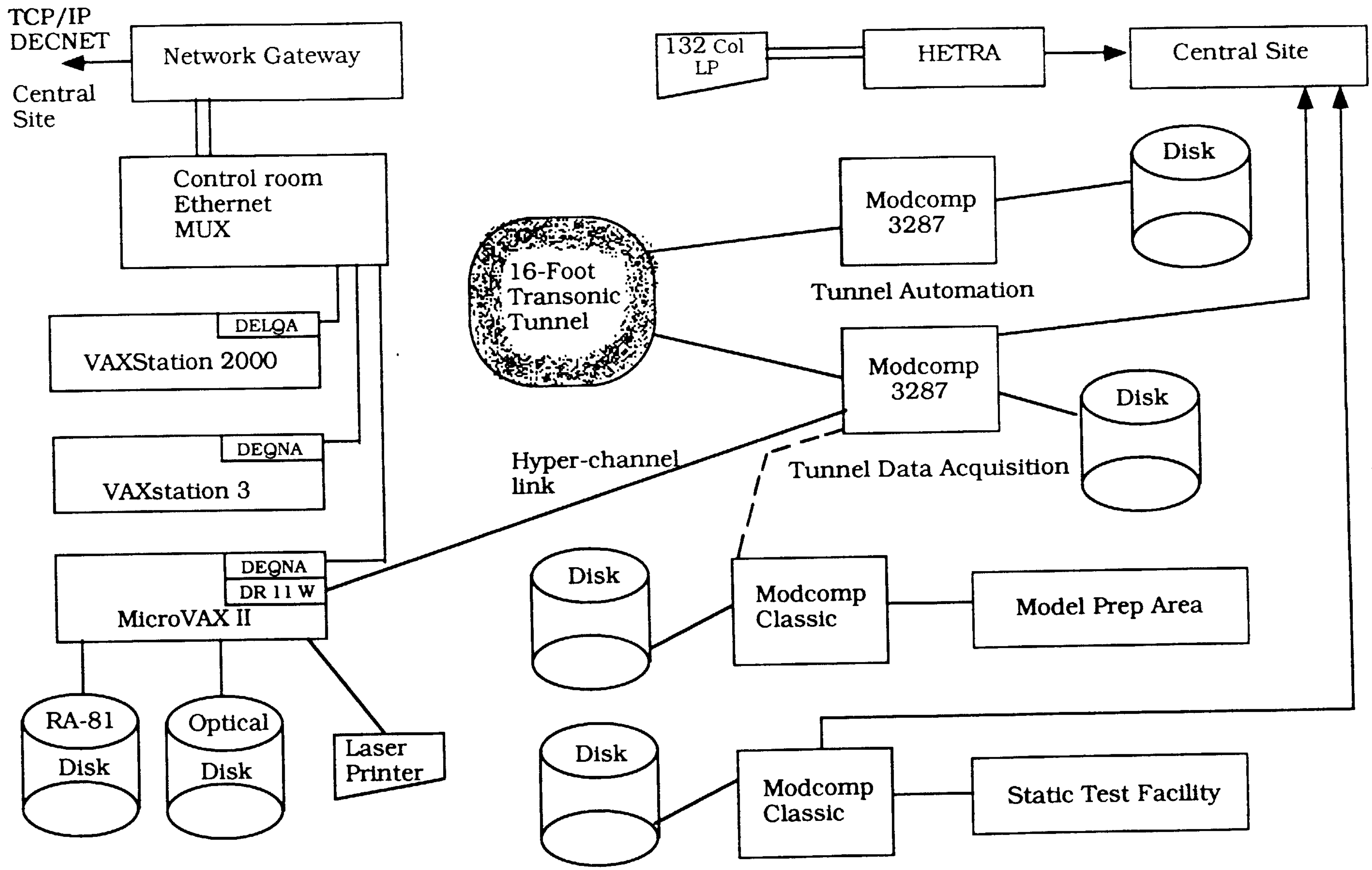

FIGURE VII-1.- Layout of 16-Foot Transonic Tunnel Complex computing hardware.

SECTION VIII - Model Design Criteria

It is the responsibility of the user to establish the model design load criteria. Documented load envelopes, sets of steady-state aerodynamic load components and thermal loads for extremes of test conditions, model configurations, and model attitudes are required. Areas to be considered in determining model design criteria include:

Model Design Requirements

Model Fabrication Requirements

Quality Assurance Requirements

Documentation Requirements

The Facility Safety Head of the 16-Foot Transonic Tunnel is authorized to implement, modify, delete, or expand the criteria listed above.

NASA/Langley Research Center Handbooks entitled, Wind Tunnel Model Systems Criteria (LHB 1710.15 dated August 1986) and Safety Regulations Covering Pressurized Systems (LHB 1710.40 dated November 1988) serve as the official requirements documents for models and model systems and are available to the user for detailed information pertaining to wind tunnel model designs.

A. Design Requirements.-

1). Design Loads.- Consideration of the following items will help establish model guidelines:

- Model aerodynamic loads
- Test temperatures
- Model scale
- Controls

- Deflections
- Jet pressure ratio range for powered models
- Static and dynamic pressures
- Instrumentation
- Balances
- Support systems

Several of these items have been discussed in other sections of this document.

2). Stress Analysis - show that allowable stresses not exceed the worst load case including:

- Dynamic factors
- Thermal stress
- Stress concentration factors

Each detailed analysis section should contain a sketch showing forces and moments acting on the part and a statement of:

- Assumptions
- Approximations
- Section properties
- Type and heat treat condition of material
- Pertinent drawing number

3). Allowable Stress - Shall be the smaller of the values of one-quarter of the ultimate tensile strength or one-third of the tensile yield strength of the material at test conditions. Oscillating stresses shall be computed as the mean stress applied to the Proper Modified Goodman Diagram to which a safety factor of four (4) has been applied.

4). Material Selection - Materials are to be selected using mechanical properties in the latest issue of one of the following standards:

- ASTM specifications
- MIL
- HDBK 5A, metallic materials and elements for aerospace vehicle structures
- MIL - HDBK - 17, plastic for flight vehicles

All mechanical properties used shall be suitably corrected for:

- Temperature
- Vacuum
- Other environmental effects which may be present when material is under stress.

5). Structural Joints - For bolt preload in bolted structural joints oscillating stresses in the threads are to be avoided and one-half of the bolt's tensile yield strength is not to be exceeded. Shear loads should be transmitted by keys, pins, pilots, or shoulders. All structural connections must be provided with positive mechanical locks. Drawings for all structural connections are to list the strength and quality of fasteners and torque values for tightening screws and nuts.

6). Pressure Systems - Models, support equipment, and test equipment using hydraulic, pneumatic, propulsion, and other systems with operating pressure, above 15 psig shall be designed, fabricated, inspected, tested, and installed to comply with the ASME Boiler and Pressure Vessel Code.

7). Support Systems - All model support systems to be furnished by the user shall meet the applicable requirements listed above. Sting designs must also satisfy the divergence criteria for the Langley 16-Foot Transonic Tunnel. That is, the slope of the model normal force variation with angle of attack must be less than 40 percent of the adjusted sting-bending restoring moment (see figure V111-1.) Note that the adjusted sting-bending restoring moment is the bending moment due to a point load which accounts for both the model normal force and pitching moment. The sting-bending restoring moment can be determined by using computational methods or preferably by actual loadings.

B. Fabrication Requirements.- The user is responsible for having models fabricated and assembled in compliance with design drawings and specifications. Models are to be assembled and discrepancies corrected before shipment. Check that:

- All model parts fit properly
- All remote controlled model components function properly
- All position indicators can be calibrated
- Sufficient clearances for deflections due to loadings
- All leads identified
- All orifices cleaned, open, and leak checked
- All propellent lines cleaned and checked at operating pressures

C. Quality Assurance Requirements.- The user shall maintain an effective inspection system which contains provisions for defining and verifying model components and material quality throughout all operations.

D. Documentation Requirements.- A model integrity report is required at least 4 weeks in advance of tunnel entry and will contain the following:

- Design loads criteria
- Stress analysis
- Drawings of model components to be tested
- Quality inspection reports to validate the integrity of the completed model
- Installation and change procedures

The procedures will contain sufficient detail, including sequential steps, torquing values, etc. to assemble, install and check out the model in the tunnel and permit model changes during the test program.

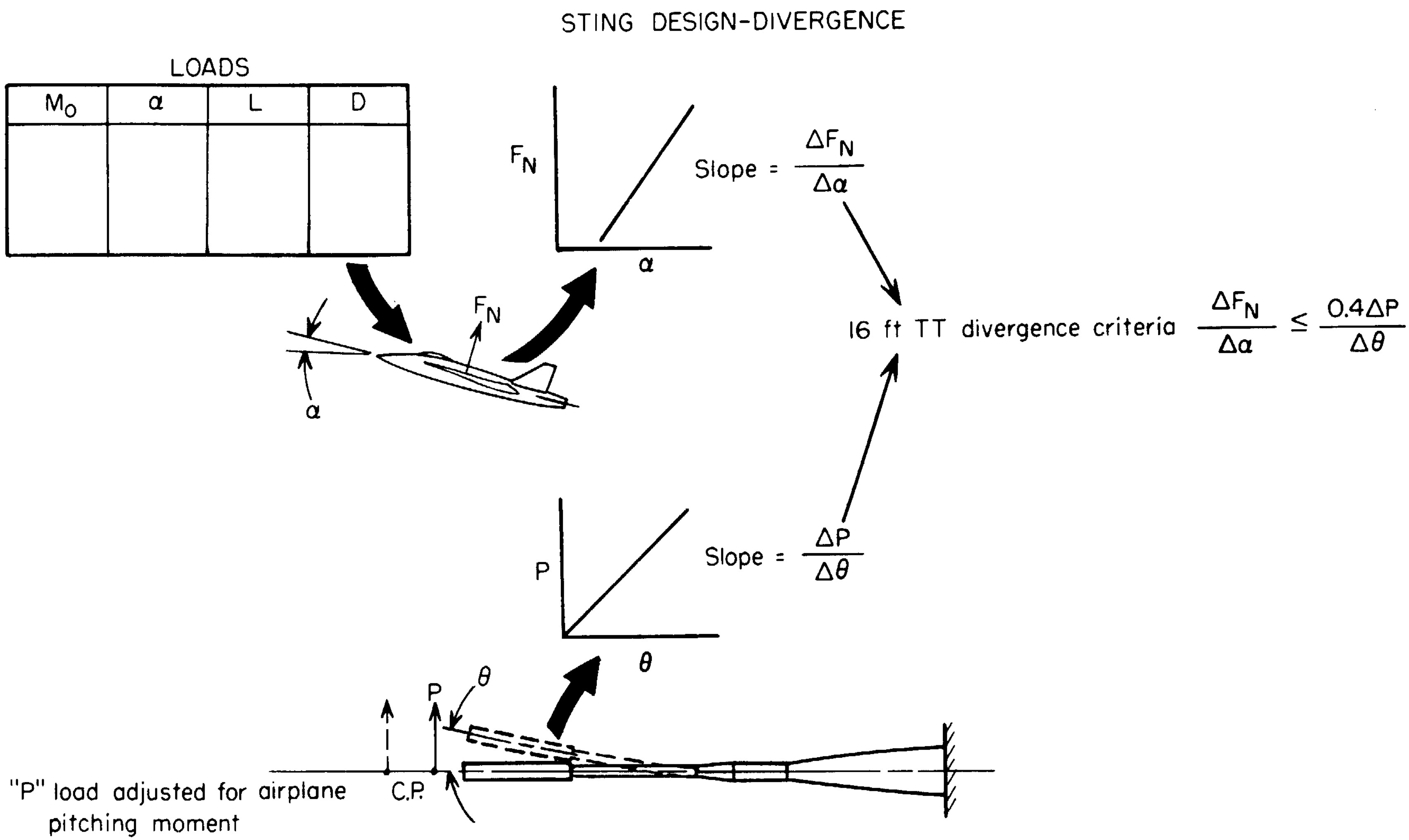

Figure VIII-1. Divergence criteria.

SECTION IX - Wind Tunnel Test Planning

It is the intention in this section of this user's guide, to inform the potential facility user of the procedures and information required to conduct a successful wind tunnel test in the Langley 16-Foot Transonic Tunnel Complex. The test planning and procedures are described as follows:

A. Initial Contact.- Potential facility users can generally establish from the published literature whether the Langley 16-Foot Transonic Tunnel has the capability of fulfilling the test program requirements. However, factors such as unique test requirements or facility modifications to accept new tests, necessitate early contact between the users and the personnel of the Propulsion Aerodynamics Branch. When the need arises, sponsors or users should contact the facility as early as possible:

NASA - Langley Research Center
Propulsion Aerodynamics Branch
Hampton, Virginia 23665-5225
Phone No.: (804) 864-3001

If Branch management is convinced that the proposed test meets the following criteria:

1. Fits into NASA, Langley, and Branch roles and missions
2. Advances state-of-the-art or is critical to a major national program
3. Is feasible (from both a technical and scheduling point of view)

a pretest planning conference will be arranged. At this point, a Memorandum of Agreement (MOA) or model loan agreement needs to be initiated. The MOA approval process is quite long and needs to be initiated as soon as possible (if an existing one is not in place). A sample (fill-in-the-blank) MOA is shown in Figure IX-1.

B. Pretest Conference.- One or more pretest conferences will be required, depending on the test program complexity, data reduction requirements, and the hardware and instrumentation requirements. At the first pretest conference, the sponsor and user should be prepared to discuss the following information:

1. Objectives of the proposed test
   - Objectives of test should supplement Propulsion Aerodynamics Branch roles and missions (i.e., inlet/nozzle performance; inlet/engine/nozzle installation effects; etc.). On rare occasions, a test outside of these roles and missions will be considered.
2. Scheduling considerations
   - When does user absolutely require data and why? It should be kept in mind that the 16-Foot Transonic Tunnel is scheduled well in advance and the schedule will be changed only under very special circumstances.
3. Test plans showing desired configurations and test conditions
   - Again, considering the heavy demand for the wind tunnel facility, test plans should be reasonable. Typical tests in the 16-Foot Transonic Tunnel generally have 2-3 weeks of actual testing (2 shifts/day; 5 days/week).
4. Instrumentation requirements
5. Data reduction requirements
6. Hardware requirements
7. Model stress analysis
   - This item is particularly important. Models must conform to the specifications detailed in Langley Handbook "*Wind-Tunnel Model Systems Criteria*" (LHB 1710.15) and be approved by the Facility Safety Head. Waivers must be obtained for each model nonconformance item. If a waiver cannot be obtained, the model will not be approved for test. If possible, this item should be addressed before model fabrication starts. This

illustrates the need for a pretest meeting as early as possible. The user should be prepared to discuss model loads and how they were (or will be) obtained.

8. Security requirements

9. MOA/loan agreement

After reviewing the above information, the Propulsion Aerodynamics Branch will conduct a detailed study of the test requirements to determine when such a test can reasonably be conducted.

C. Documentation Requirements.- A pretest report prepared by the users should be delivered to the designated NASA Project Engineer at a time mutually agreed to by the two parties. This package should include, but not be limited to the following:

1. Complete set of "as-built" model drawings
2. Complete set of drawings showing propulsion system and any other instrumentation requirements.
3. Model stress report
4. Calibrations of user furnished instrumentation and propulsion equipment when appropriate
5. Detailed test plan describing model configurations and test matrix
6. Data reduction requirements
7. Any special operating procedures

D. Model Delivery and Inspection.- Models should be delivered to the Langley 16-Foot Transonic Tunnel at least 3 weeks prior to the scheduled starting date of the wind tunnel test. When special instrumentation is necessary or when calibrations are required, for example, calibrating electrical strain gage balances, additional lead time will be required. User-supplied electrical strain gage balances should be delivered to NASA at least

7 weeks prior to the scheduled starting date of the test (4 weeks for calibration; 3 weeks for model build-up and tares). In all cases, the model will be thoroughly inspected by NASA personnel before the model is put into the tunnel. The detailed requirements for this inspection will be identified during the pretest conference.

E. Test Operations.- The project engineer will determine, with the aid of the user, the number of user personnel involved in the test and the interaction between these persons and the Langley support staff during all phases of the test program from model preparation to data processing. In all cases, however, the user will have on site, during all phases of the program, at least one person who is thoroughly familiar with the model. The NASA project engineer is totally responsible for the successful completion of the test. In that regard, the user's representative must communicate with the test personnel through this engineer.

F. Final Data Package.- The Langley 16-Foot Transonic Tunnel has a completely automated data acquisition and reduction system. In many cases, near final data can be obtained within a few minutes after the completion of the test. In most cases, however, there are corrections which must be applied to the data that cannot be obtained prior to the initiation of the test. Generally, a final set of data will be transmitted to the user and/or sponsor within 30 days of the completion of the test.

G. Report Requirements.- It is the policy of the Langley Propulsion Aerodynamics Branch that all data obtained in the Langley 16-Foot Transonic Tunnel be published in NASA documents. However, under certain highly special circumstances this policy may be waived. The exact methods of publication of these data are the responsibility of the project engineer. In all cases, the user and/or sponsor will decide upon their own publication methods.

MEMORANDUM OF AGREEMENT

FOR

BETWEEN

THE NATIONAL AERONAUTICS AND SPACE ADMINISTRATION

LANGLEY RESEARCH CENTER

AND

The National Aeronautics and Space Administration, Langley Research Center, (hereinafter referred to as NASA/LaRC), acting under the authority of the National Aeronautics and Space Act of 1958, as amended, and [name of company] (hereinafter referred to as __________), jointly execute this Memorandum of Agreement (MOA) in order to [Describe mutually beneficial cooperative program/study].

1.0 PURPOSE

The parties hereto agree that: [Describe nature of cooperative effort/study/testing and objective(s) to be obtained/data to be collected.]

2.0 RESPONSIBILITIES OF THE PARTIES

2.1 NASA/LaRC hereby agrees to use its best efforts to:

Figure IX-1. Sample MOA.

2.1.1

etc.

2.2 [Name of company] hereby agrees to:

2.2.1

etc.

3.0 DATA, DATA RIGHTS AND PUBLICATION

Both parties have equal rights to all data collected pursuant to this MOA, and there are no restrictions or limitations on data use or publication by either party.

4.0 PERIOD OF AGREEMENT

This MOA shall remain in effect for a period of ______ months(s)/year(s) from the latest date upon which all parties have executed same.

5.0 COSTS AND FUNDING

There will be no transfer of funds or other financial obligation between NASA/LaRC and [name of company] in connection with this MOA. NASA's performance of this MOA is subject to the availability of NASA's appropriations therefor, and further, nothing expressed herein commits the United States Congress to appropriate funds therefor. However, NASA/LaRC agrees to use its best efforts to obtain the necessary funding.

Figure IX-1. Continued.

6.0 LIABILITY

6.1 NASA and [name of company] agree that with respect to injury, death or damage to persons or property involved in operations undertaken pursuant to this MOA, neither NASA nor [name of company] shall make any claim with respect to injury or death of its own or its contractors' or its subcontractors' employees or damage to its own or its contractors' or subcontractors' property caused by activities arising out of or connected with this MOA, whether such injury, death or damage arises through negligence or otherwise.

6.2 [Name of company] shall not make any claim against the United States Government (or its contractors or subcontractors) for damages or other relief for any delay (including a deferral, suspension or postponement) in the provision of any service under this MOA or for the nonperformance or improper performance of such services, including, but not limited to the performance by the United States Government or by the United States Government's contractors and subcontractors.

7.0 INDEPENDENT RELATIONSHIP

This MOA is not intended to create, constitute, give the effect of or otherwise recognize a joint venture, partnership, agency or formal business organization of any kind, and the rights and obligations of the parties hereto shall be only those expressly set forth herein.

Figure IX-1. Continued.

8.0 UNITED STATES GOVERNMENT OFFICIALS NOT TO BENEFIT

No member of or delegate to the United States Congress, or resident commissioner, shall be admitted to any share or part of this MOA, or to any benefit that might arise therefrom, but this provision shall not be construed to extend to this MOA if made with a corporation for its general benefit.

9.0 MODIFICATION AND TERMINATION

This MOA sets forth the entire and complete agreement between NASA/LaRC and [name of company] and may be modified only by written mutual agreement. This MOA may be terminated prior to its expiration by either of the parties upon written notice to the other party communicated not less than sixty (60) days in advance.

______________________________ [Name and title of individual authorized to execute MOA]	______________________ Date
______________________________ [Division Chief] NATIONAL AERONAUTICS AND SPACE ADMINISTRATION LANGLEY RESEARCH CENTER	______________________ Date

Figure IX-1. Concluded.

SECTION X - Static Test Facility

In conjunction with the propulsion integration efforts of the 16-Foot Transonic Tunnel, scale model nozzle internal performance research is necessary to evaluate new or novel concepts. Some of these tests are in direct support of tunnel experimental work while others are on model hardware designed only for nozzle internal performance (static) evaluation with no direct link to a specific wind tunnel program. Nozzles intended for static testing only, can be fabricated more economically since there is no need to represent external model geometry. These tests are conducted in the Static Test Facility (Building 1234) which is on the same site as the 16-Foot Transonic Tunnel and is under the organizational control of the tunnel staff.

The propulsive flow medium is dry high-pressure air exhausting to ambient (atmospheric) back pressure within the building. The high-pressure air which is continuously supplied from a central 5000 psi system is expanded into an 1800 psi reservoir outside the building and then regulated inside the building as required for a particular test condition. Within the facility, a portion of the high-pressure air is passed through a steam heat exchanger by a regulator to maintain the nozzle exhaust flow at a selected constant temperature between 65°F and 90°F. The flow rate of the present high-pressure air system is 15 lbs/sec and is metered by critical flow venturis. Installation of an additional high-pressure air system along with a planned upgrade of the present air system is in the planning stage so that in the future two separate flows, each having the capacity of 30 lbs/sec, will be available.

The Static Test Facility has its own on-site data acquisition system (NEFF) and computer (MODCOMP Classic). An electronic pressure scanning unit (32 port ESP) is available and most of the portable instrumentation (e.g. individual pressure transducers) available to the tunnel can be used in the Static Test Facility. The data reduction procedures for static tests are part of the generalized data reduction package used by the 16-Foot Transonic Tunnel so that a wide range of computed parameters are available without reprogramming. An acoustically triggered schlieren system is available to study shock structure in nozzle exhaust plumes and a portable three component laser doppler velocimeter should be available by the end of 1990.

Access to the static test bay from outside the building is provided by two large doors opening to a driveway so that the static test stand can be removed and replaced with some other test apparatus. The jet exhaust is vented to the outside of the building through a louvred vent in the roof so that the test bay can be secured. The building housing the Static Test Facility is shared with a water tunnel (Section XI) which is in an adjacent room and is also under the organizational control of the 16-Foot Transonic Tunnel staff.

A. Single-Flow Propulsion Simulation System.- The single-flow propulsion simulation hardware described in Section IV-A is the same as the hardware used in the Static Test Facilty. Several sets of components exist and can be assembled to the various support systems. The single-flow system (shown schematically in figure X-1) is usually mounted on a wheeled dolly-supported strut as shown in figure X-2 when used for nozzles designed for static-only testing. The conventional static test installation uses the flanged low-pressure plenum mounted to the bellows-force balance assembly with existing round or round-to-rectangular instrumentation sections attached. The round instrumentation section (shown in figure X-3) is available for testing axisymmetric nozzles while the round-to-rectangular instrumentation section (shown in figure X-4) is available for testing nonaxisymmetric nozzles. Other low-pressure plenums or instrumentation sections can be built as required for specific needs. Calibration of the force measuring system (force restraints and momentum tares) and verification of the venturi flow measurements is conducted with the same Stratford choke nozzles installed (figure V-1) as are used in the tunnel. The largest capacity force balance currently available for use with the single-flow hardware shown in figure X-1 is the 1631 balance (figure VI-8). The approximate range of nozzle pressure ratios attainable for the single-flow propulsion system (15 lbs/sec) is shown in figure X-5 as a function of nozzle throat area.

B. Dual-Flow Propulsion Simulation Systems.- The air supply for the dual-flow propulsion simulation system shown in figure X-6 is an interim system which will eventually be upgraded to a system having two independent air sources. It can be seen in this figure that the primary and secondary air supplies are obtained from a single air supply. This air supply is capable of providing 15 lb/sec of clean, dry air at 560°R. After the flow is split, the pressure and flow rate of the primary and secondary air supplies can be regulated and monitored (thermocouples and venturi flow meters) independently. Also, as shown in this figure, a Y-joint is provided for alternate operation; i.e. low flow through one venturi flow meter. If required, the primary and secondary air supply functions can be switched at the connection to the flow system on the support dolly. In the future, addition of upstream equipment will provide 30 lb/sec in each flow system.

Figure X-7 depicts the calibration and experimental set-up. Primary and secondary air supplies enter the dual-flow system assembly at rigid connectors on the support dolly. From these, the air flows through flexible stainless steel S-shaped tubes to the metric portion of the dual-flow system assembly. The function of these tubes is to provide flexible connections capable of maintaining a calibration, while transferring air to the portion (metric) of the flow system assembly whose forces and moments are measured by the six-component strain gage balance. Calibration fixtures for all six components (normal force, axial force, pitching moment, rolling moment, yawing moment, and side force) are an integral part of the support dolly. Cables and pulleys are used to position the calibration weight vectors through the required loading points for balance calibration. The expanded metal shield prevents the air supply tubes from being inadvertently moved once the system has been calibrated. Jacks are used to level the balance in the horizontal planes and the support dolly is secured to fixed attachments in the facility concrete floor by cables.

The dual-flow system assembly is shown in Figure X-8. The secondary flow system, consisting of a concentric outer structure, can be removed if required. The NASA 1635 strain gage balance, was used in this assembly and has the following capacity:

normal force = 600 lb

axial force = 1,200 lb

pitching movement = 12,000 lb

rolling moment = 1,000 in-lb

yawing moment = 12,000 in. lb

side force = 600 lb.

(See section VI, figure VI -10 for balance geometry.)

The accessory components shown in Figure X-9 provide means by which the primary and secondary flows can be isolated into a single flow for calibration using the Stratford choke nozzle of figure V-1 or for experiment.

Figure X-10 illustrates a type of dual flow nozzle model that can be tested, using the dual-flow propulsion simulation system. The model displayed in this figure is a vectoring axisymmetric ejector nozzle whose geometric variables are vector angle and top and bottom ejector gap height.

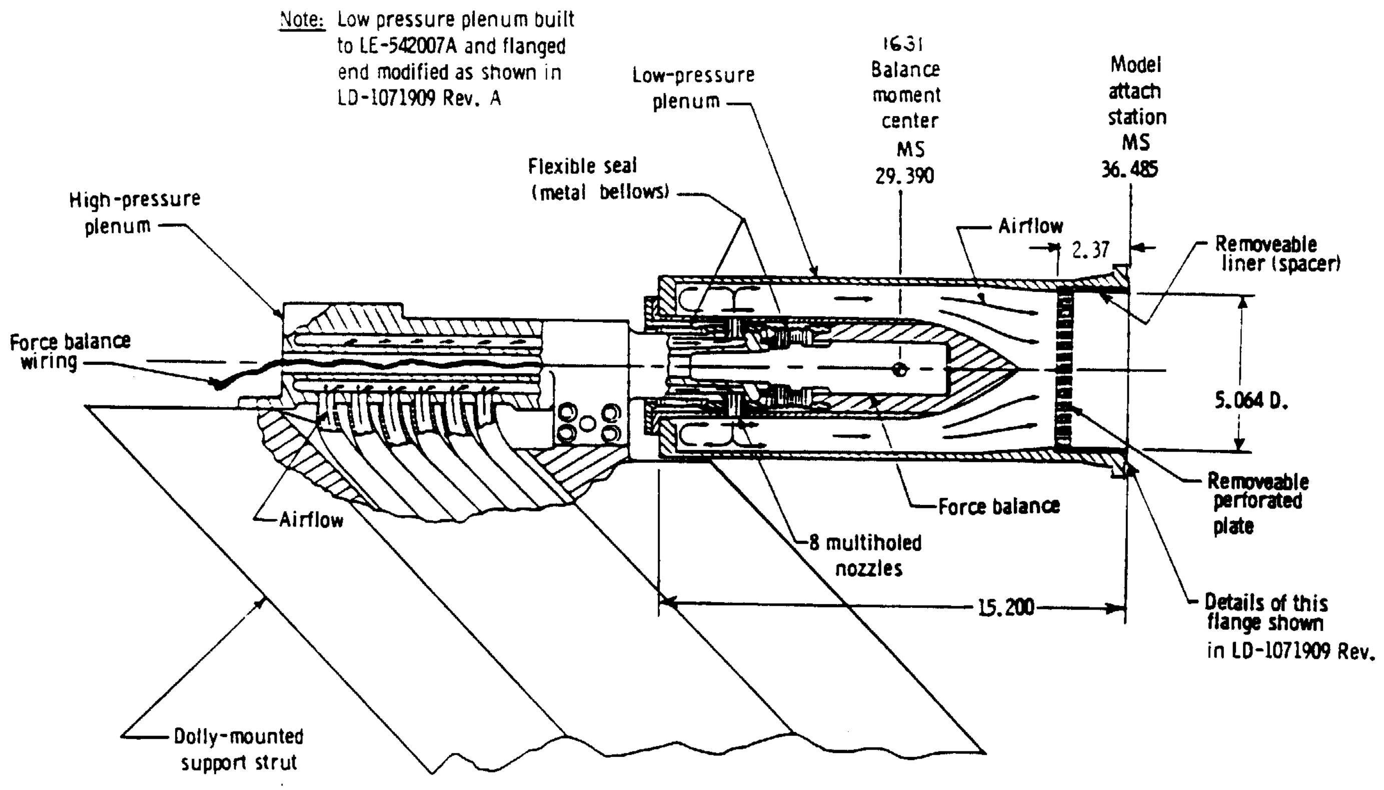

Figure X-1. Schematic of flanged single engine propulsion simulation test pod. (All dimensions are in inches.)

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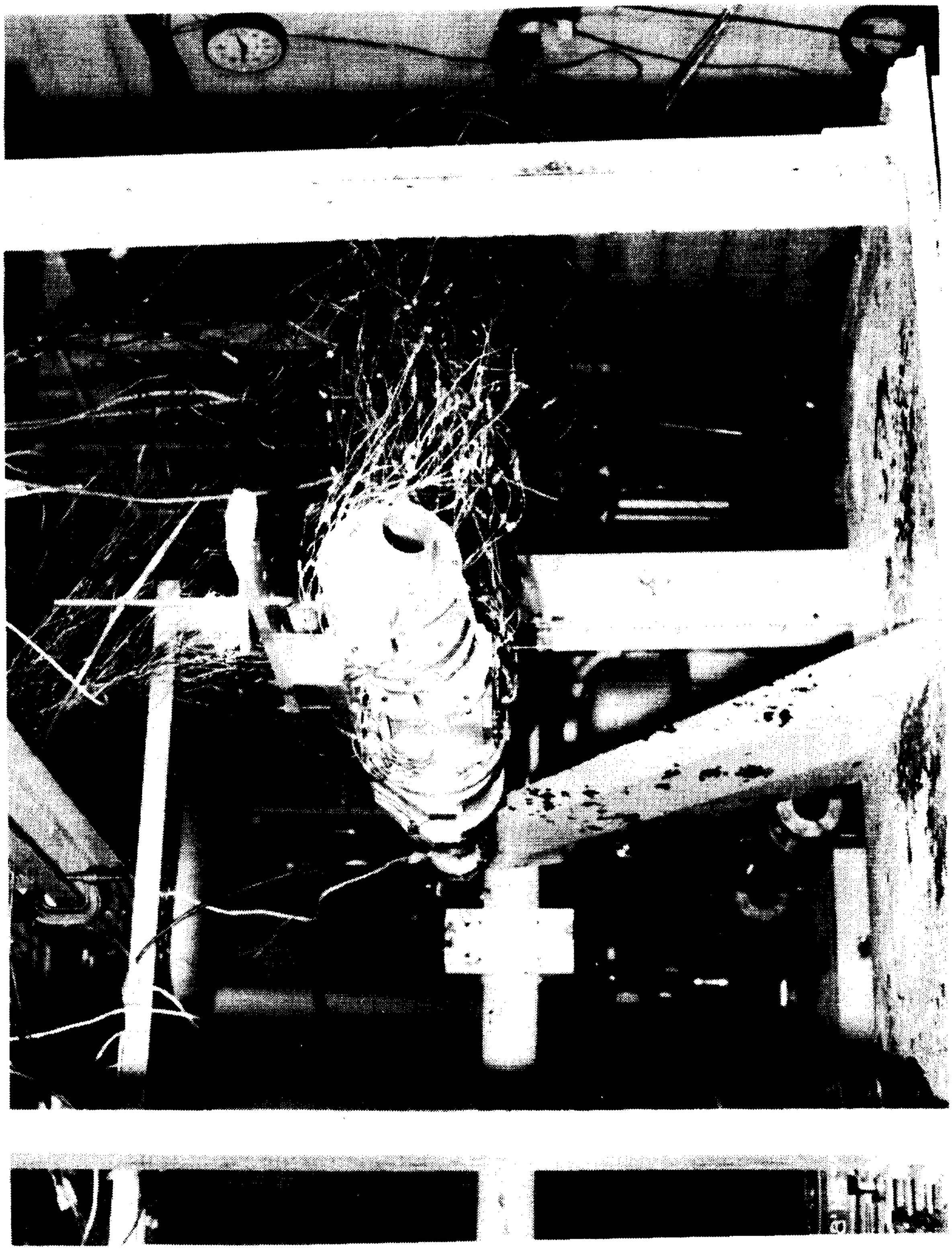

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Figure X-2. Single-flow propulsion hardware with a vectored axisymmetric nozzle mounted on the static test stand strut.

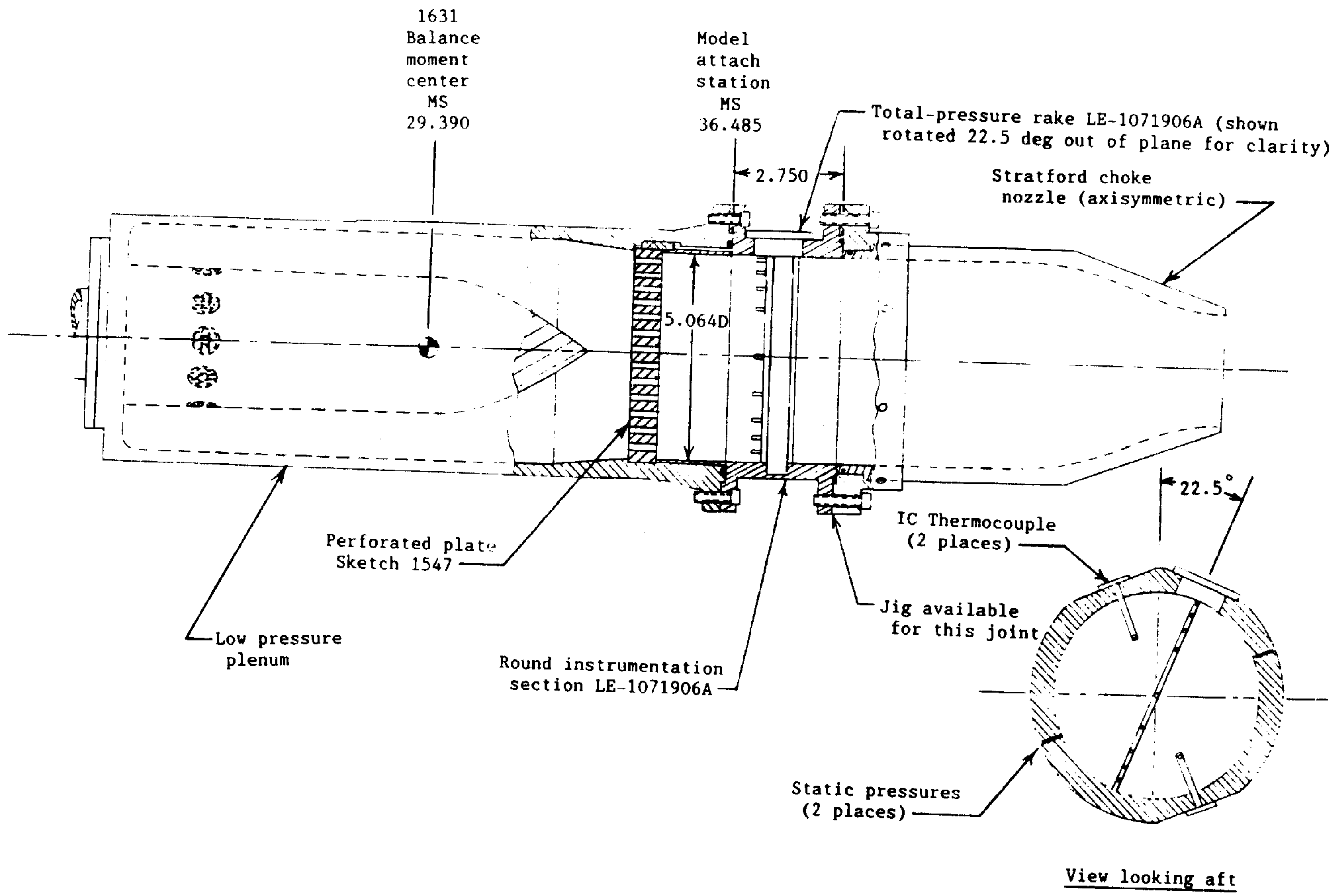

Figure X-3. Sketch of single-flow static test hardware assembly with round instrumentation section installed. (All dimensions are in inches unless otherwise indicated.)

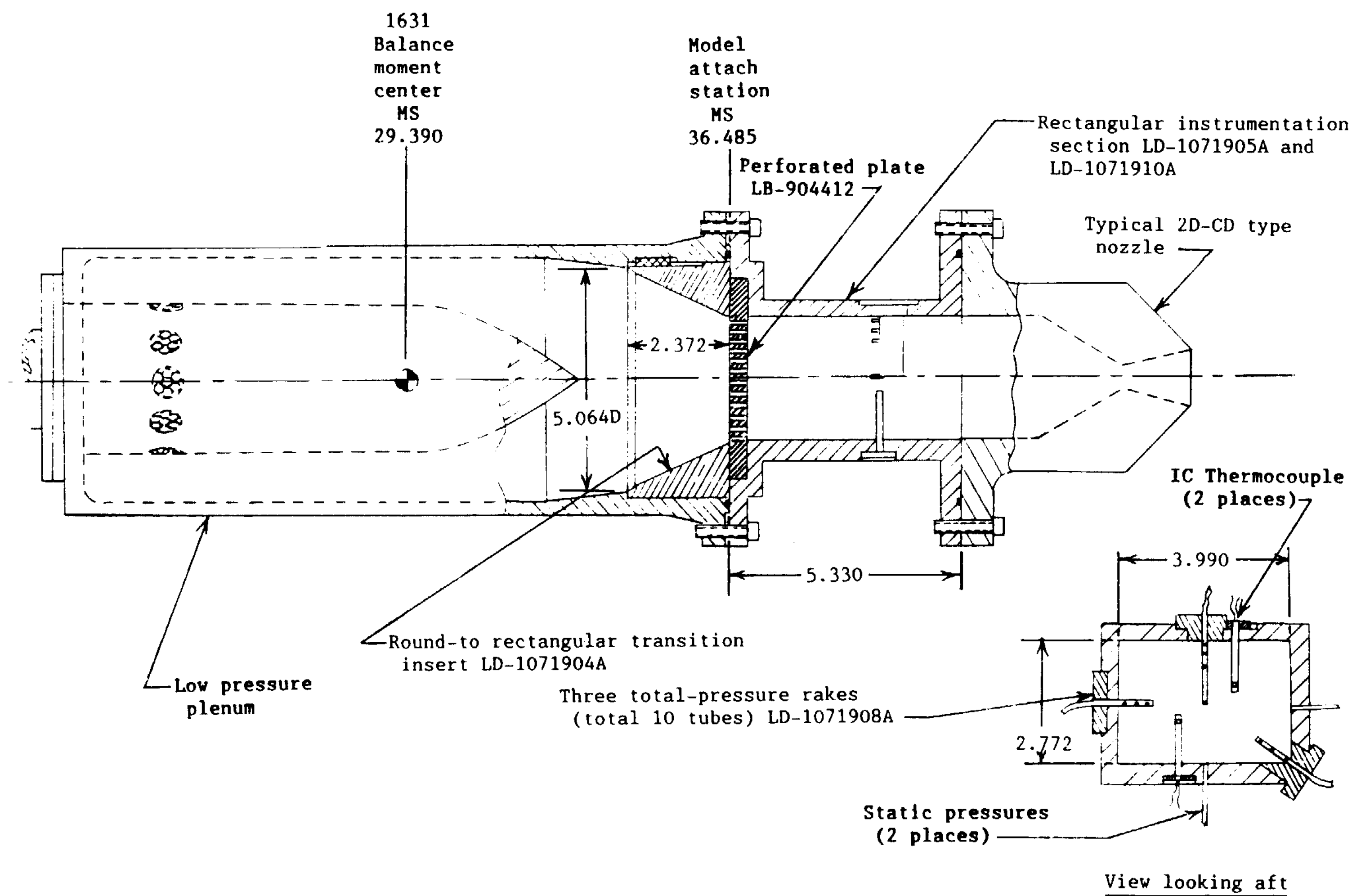

Figure X-4. Single-flow static test hardware assembly with rectangular instrumentation section installed. (All dimensions are in inches unless otherwise indicated.)

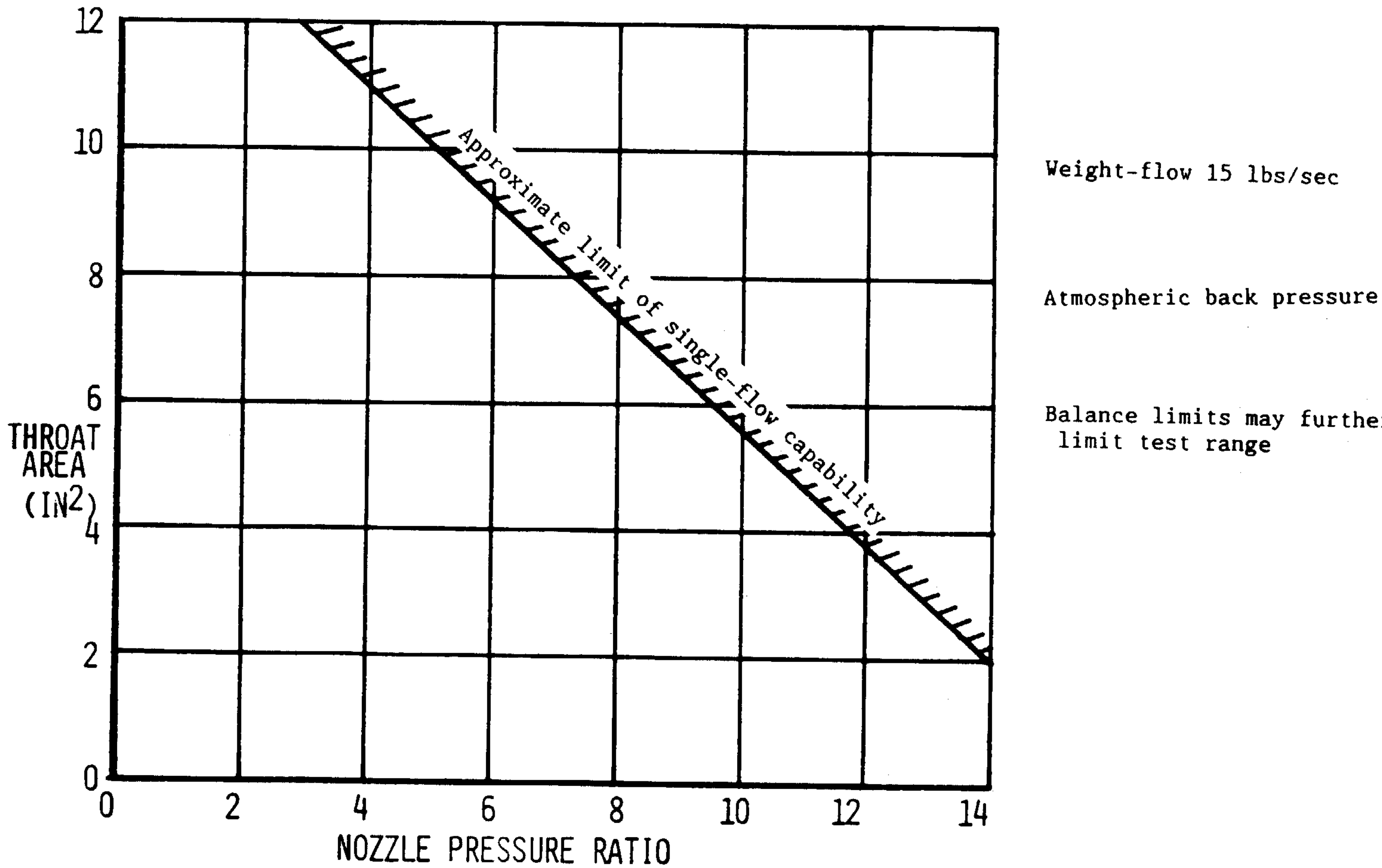

Figure X-5. Approximate range of nozzle pressure ratios attainable for the single-flow propulsion simulation system as a function of nozzle pressure ratio.

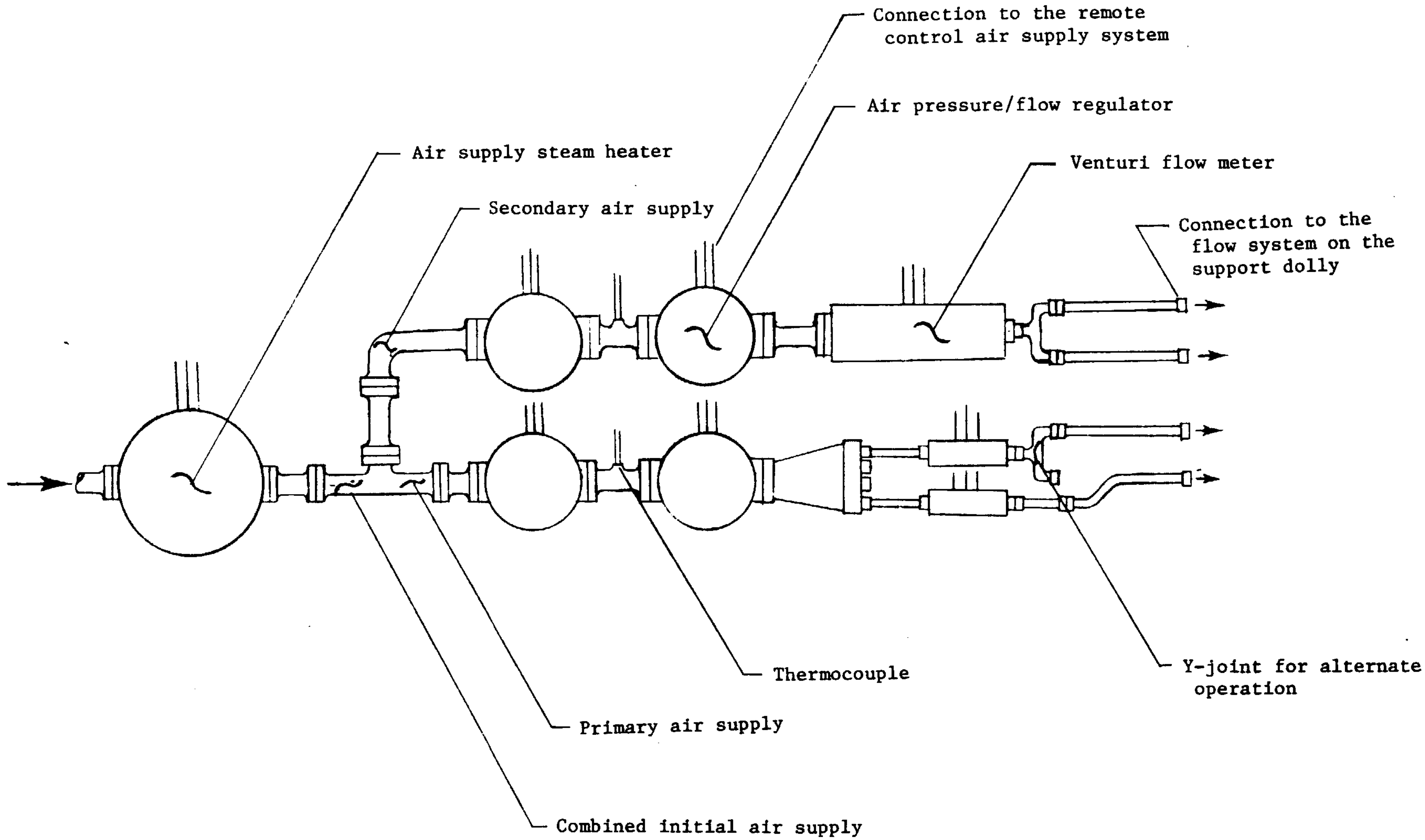

Figure X-6. Dual-Flow Propulsion Simulation System, air supply schematic.

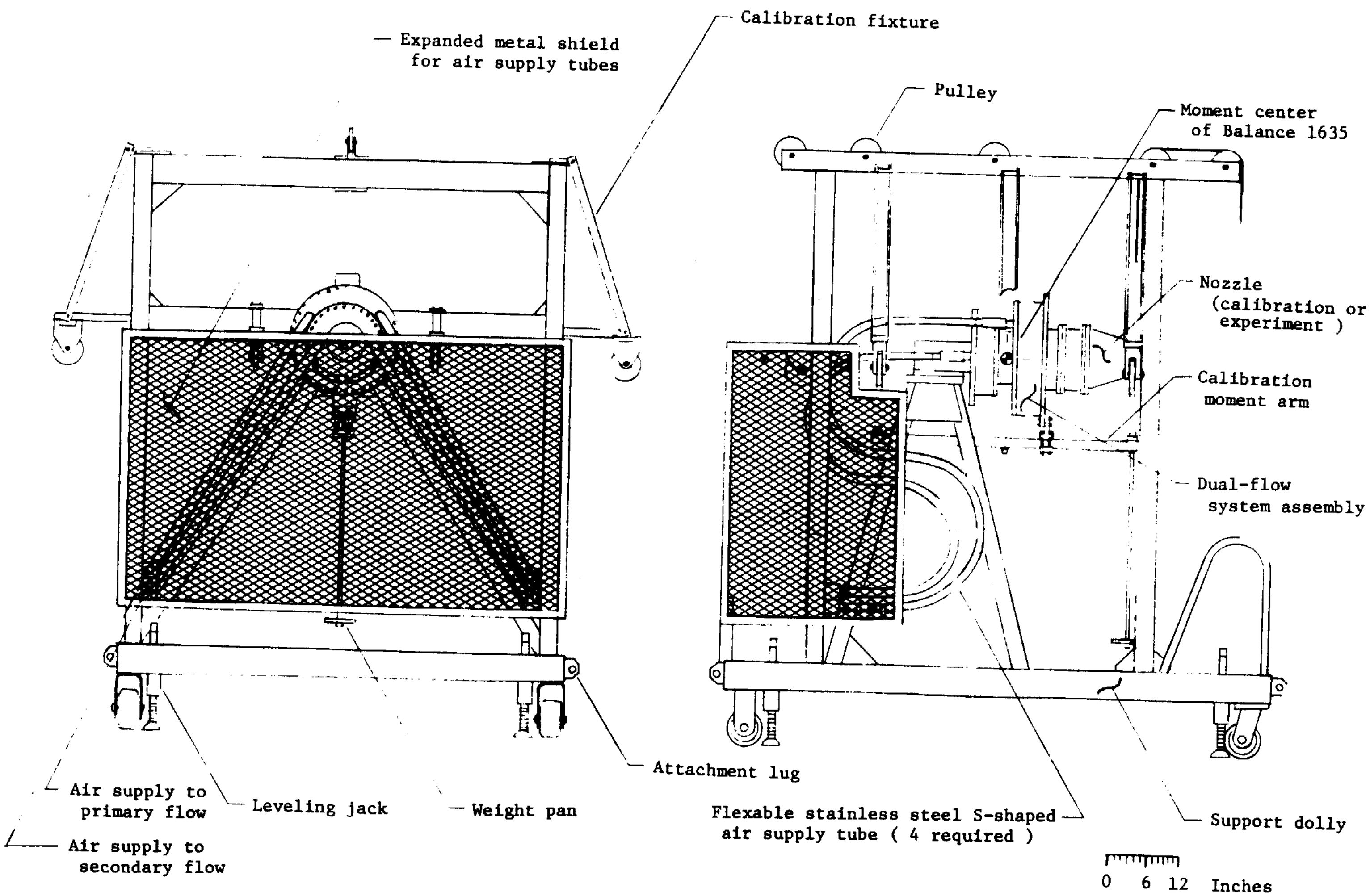

Figure X-7. Dual-Flow Propulsion Simulation System, calibration and experiment set-up.

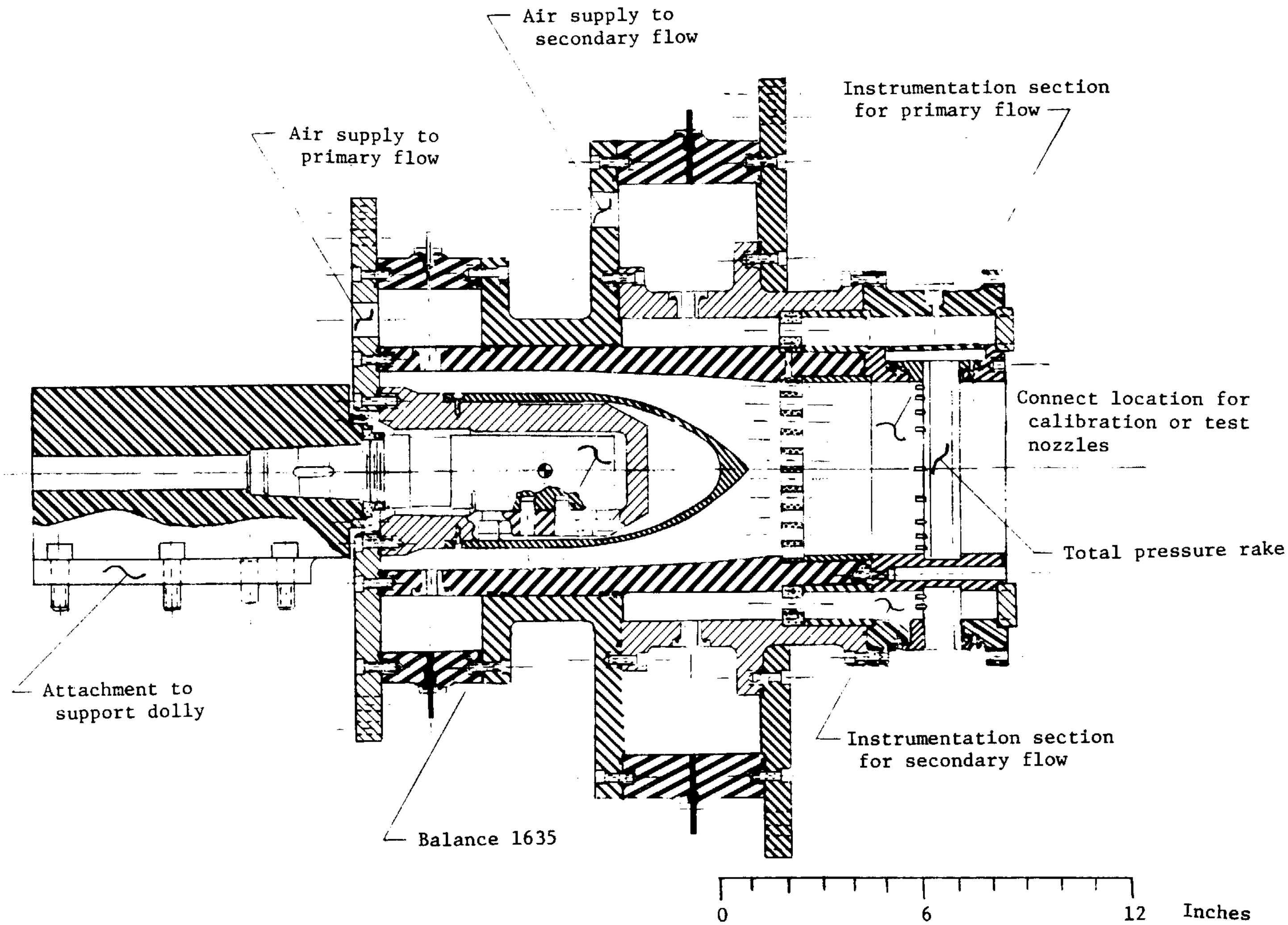

Figure X-8. Dual-Flow Propulsion Simulation system, flow system assembly.

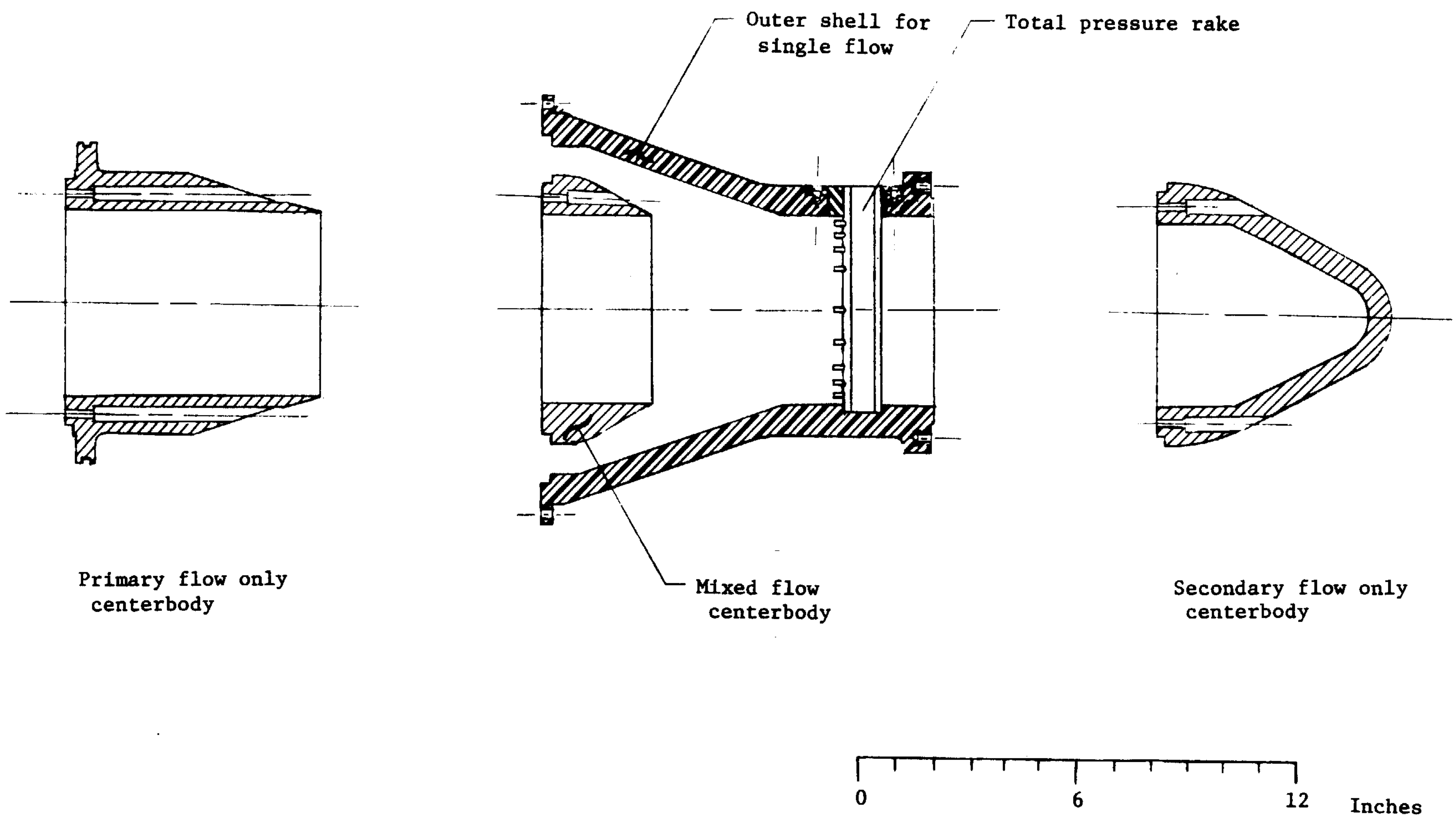

Figure X-9. Dual-Flow Propulsion Simulation System, accessory components.

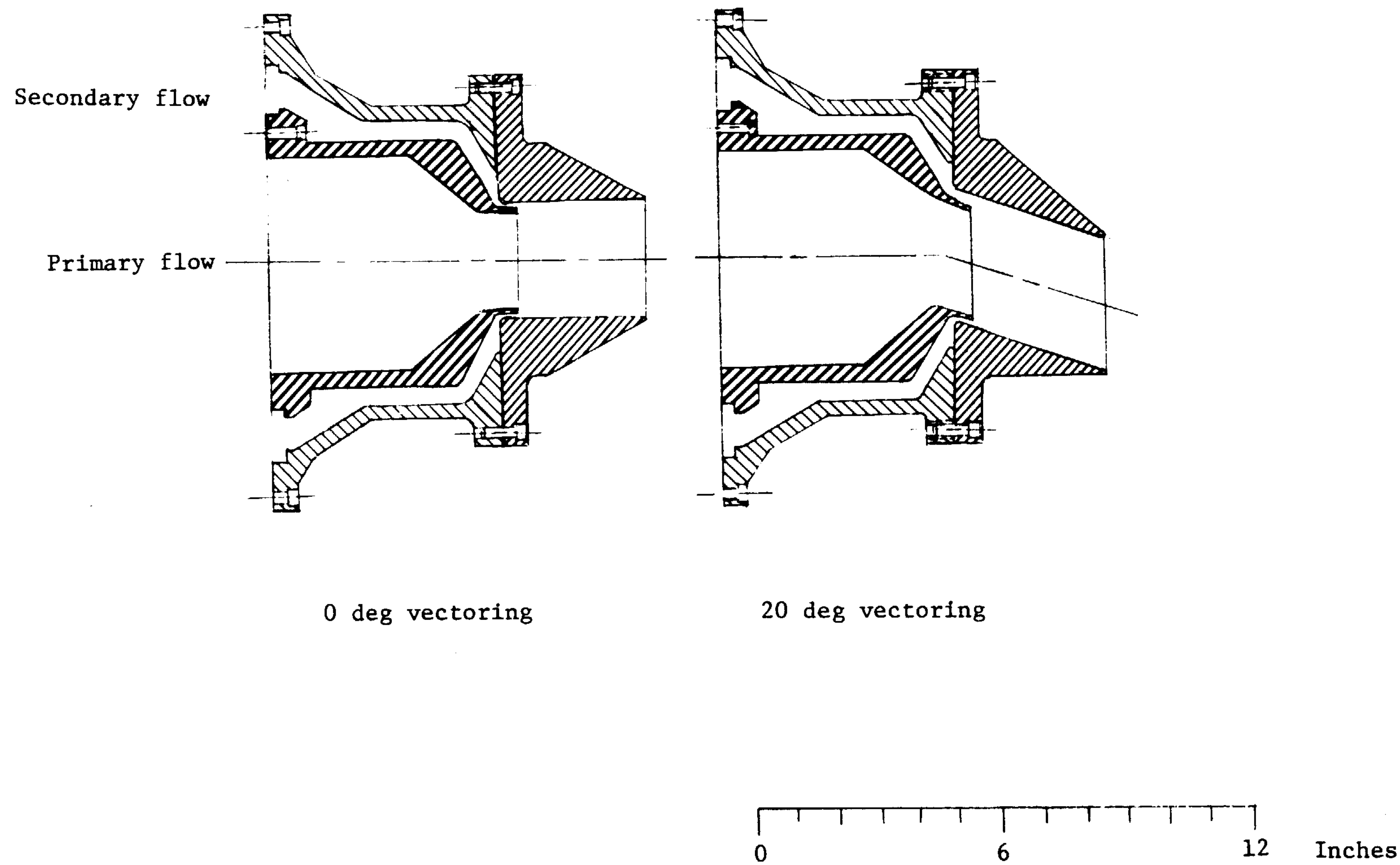

Figure X-10. Dual-Flow Propulsion Simulation System, Vectoring Axisymmetric Ejector Model.

SECTION XI - Langley 16- by 24-Inch Water Tunnel

Water tunnels are used to visualize Reynolds number independent flow patterns around sealed flight vehicle models and other specific aircraft components used to create particular flow patterns. Water tunnels provide a quick look at flow characteristics around models which can easily and inexpensively be modified to explore new designs before they reach more expensive wind tunnel test phases. These tunnels have been growing in popularity because of their low-cost operation, their ability to generate three-dimensional flow patterns, that can be easily photographed, and the ability to rapidly change models or model shapes.

A. Description of Facility. - The Langley 16- by 24-Inch Water Tunnel is located in Building 1234 at the Northwest corner of the Langley 16-Foot Transonic Tunnel. The tunnel was designed in 1983 and construction took 2 years with completion in April 1986. After a test section access door failure in May 1985, and subsequent redesign and reconstruction of the test section and door, with external corner braces added, the tunnel began operational use on March 28, 1988. This facility is a closed-return water tunnel with a clear plexiglas 16- by 24-inch by 6 feet 4 inch long test section with downward vertical flow capable of speeds from 0 to approximately 3/4 feet per second. The test section is located so that the model is about eye level above the floor--convenient for photographic purposes. An isometric sketch of the entire tunnel assembly is shown in figure XI-1 with the important features indicated by arrows. Figure XI-2 is a photograph taken from near the stairs looking toward the test section. The 3 dye reservoirs and the dye flow/overhead probe control console is shown to the right of the test section; and the traversing probe mechanism can be seen above the top edge of the tank over the test section. Vegetable dyes are injected into the stream flow through a remotely controlled probe mounted from above the test section (mentioned above) and/or orifices located on the model surfaces, to obtain video and/or photographic records of the flow patterns around the model. Vortex flows are especially suited to this method since the vortex core tends to contain the dye stream, if properly located.

A sketch of the angle-of-attack/angle-of-yaw mechanism is shown in figure XI-3 to show the method of attachment for models where angles must be varied remotely during the test runs. With the AOA mechanism in place, the test section is shortened about 2 feet. Models may be mounted as shown to get ±33° angle-of-attack with ±15° of yaw. If desired, model can be rotated 90° in the sting attaching fixture to get ±33° in the yaw direction. An offset sting-mounted plate is also available for mounting larger semispan wings where tip vortex flow studies are desired. Figure XI-4 shows a photograph of such an installation. The test subject is a straight wing with serrated trailing edge. The sting/strut connected to the AOA mechanism is visible at the bottom of the photo, but the 2 inch wide by 1/8 inch thick by 24 inch long wing support plate is hidden behind the test section corner brace at left in the picture. Dye was injected into the flow from the overhead traversing probe. If the AOA mechanism is not needed for the tests to be performed, then the model may be mounted directly to the support plate stand offs, however, this is recommended only when full test section length is required. Figure XI-5 is a sketch which outlines the inside surface of the test section and shows the other AOA mechanism hardware that is attached to the access door, and the yaw are support rail mounted inside the test section opposite the access door. Figure XI-6 is a photograph of the test section access door resting on the work dolly next to the test section opening. The AOA mechanism is shown installed on the door supporting an arrow wing model.

A summary of important water tunnel characteristics is given below:

LANGLEY 16" X 24" WATER TUNNEL CHARACTERISTICS

Tank Size:	25 feet long, 15 feet high, 3 feet wide
System Water Capacity:	5,250 gallons
Entrance Nozzle:	Rectangular cross section Entrance area 1440 square inches Exit area 384 square inches Length 18.5 inches (variable) Contraction ratio 3.75:1
Test Section Dimensions:	16" X 24" X 54"
Flow Velocity Range:	0 to 0.75 ft./sec.
Normal Test Velocity:	0.25 ft./sec. (300 gal./min.)
Angle of Attack Range:	-33° to +33°
Angle of Sideslip Range:	-15° to +15°

B. Flow Visualization and Recording Methods. - For most applications, the use of vegetable dyes will work best. The panel containing the 3 dye reservoirs is shown in figure XI-2 and normally uses red, green, and blue dyes, since these are the only colors with sufficient contrast and color difference for suitable photography. Dye flow rates are controlled by manually adjusting needle valves near each of the 3 reservoirs, and shut-off solenoid valves are remotely controlled from the control console. To eliminate holes, and hence potential leaks, in the test section, the 3 dye tubes run up to the top lip of the tank and either connect to the overhead probe above the model, or run down inside to the bottom of the test section where they are connected to the model. When multiple orifice tubes are used, they can be manifolded together (about 6 to 10 maximum - each) on each reservoir in any order desired. When using vegetable dyes, 3400°K flood lights are available for use at the water tunnel but should not be placed closer than 3 feet to the test section. If other types of dyes such as fluorescent dyes are necessary for the investigation, they may require a safety permit issued by the Langley Safety Engineering Branch. If illumination by laser light sheet is desired, this will also require a safety permit, plus the possible installation of door interlocks before the tests, to prevent accidental exposure of the eyes and skin to the laser output. For further information on the above safety requirements contact the 16-Foot Transonic Tunnel Safety Head.

For recording purposes, a Hasselblad 2 1/4-by 2 1/4-inch frame size, 500 EL/M motorized camera with 12 exposure and 70 exposure, 70 mm film magazines is also available on site. There are enough different lenses and accessories so that most any type view of the test subject can be made. There are, of course, limitations imposed by the test section size and external braces, along with lighting limits, but these rarely cause serious difficulties. Still photography using 70 mm film produces very clear photographs that may be enlarged, but of course, do not show movement. If this is important, using a video camera to produce a record of the movement may be necessary. Video cameras also require much less light than film cameras, but the images are not as clear. A video camera and recorder are also available on site for this purpose.

C. Test Model Requirements. - Models must be constructed of materials impervious to water and not subject to corrosion due to extended immersion in high mineral content water. Adhesives used to assemble model parts must also meet the above requirements. Model surfaces should be smooth and painted white to show dye flow filaments to best advantage. For mounting on the AOA mechanism, the model should incorporate a sting extending about 4 inches aft of the model that is slightly less than 1/2 inch in diameter (.497 to .498 inch dia.). Model size should be no larger than approximately 15 inches long or about 50 in^2 wing area if it is a conventional aircraft with fuselage, wings and tail surfaces. Models may be built with flowing inlets that are attached to exhaust tubes at the rear of model about 1/2" inside diameter.

The above description of the water tunnel characteristics and additional equipment necessary for its use should serve as a guide to the reader to determine if the water tunnel can meet his needs. If so, contact should be made with the Propulsion Aerodynamics Branch personnel for further information about the tunnel and model requirements and construction techniques as well as schedule planning for the tests. Additional information about the water tunnel and its use will be available from "A User's Guide to the Langley 16- by 25-Inch Water Tunnel" to be published as a NASA TM.

SKETCH OF LANGLEY 16 BY 24 INCH WATER TUNNEL

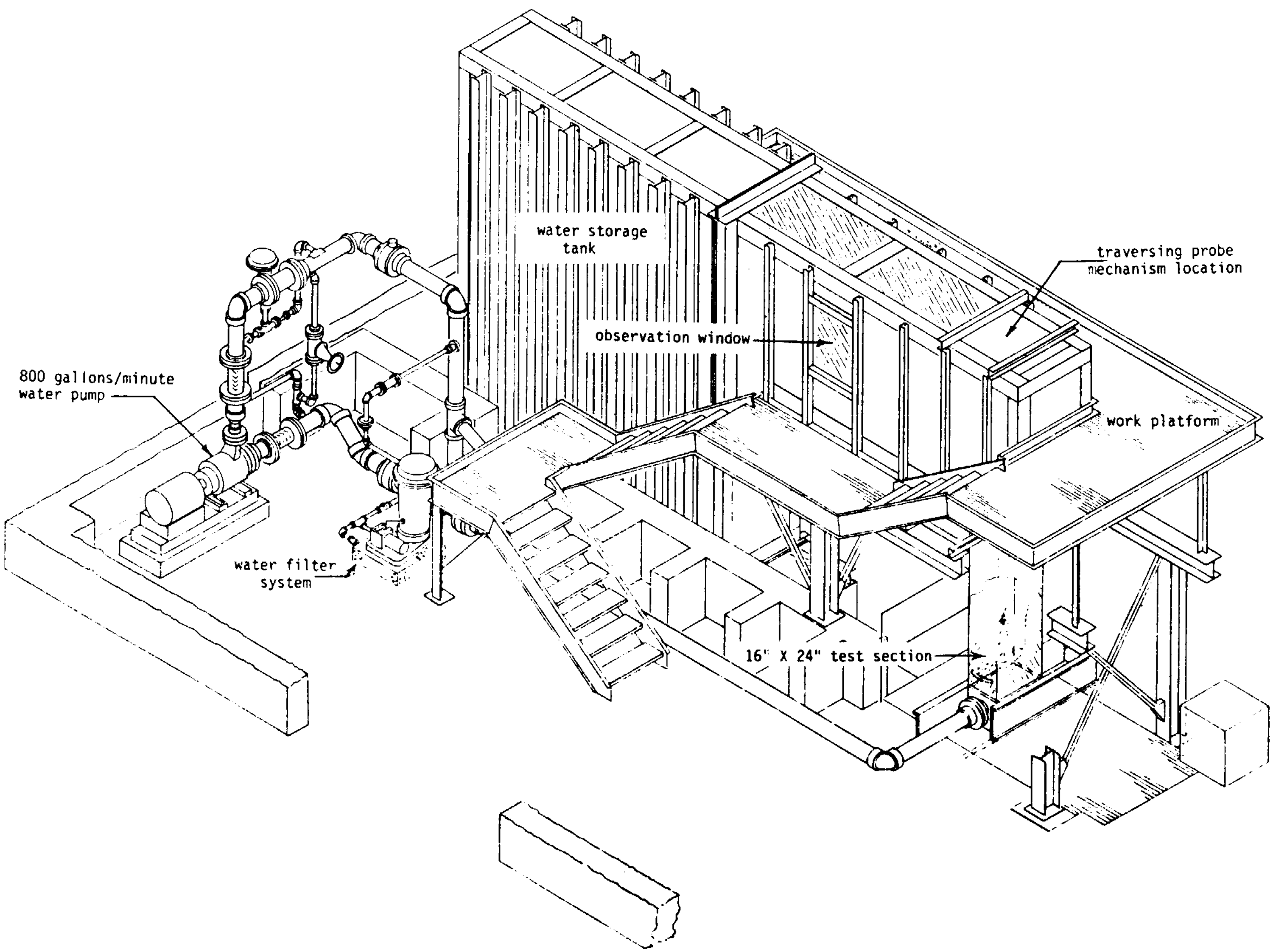

Figure XI-1. Sketch of water tunnel showing important features.

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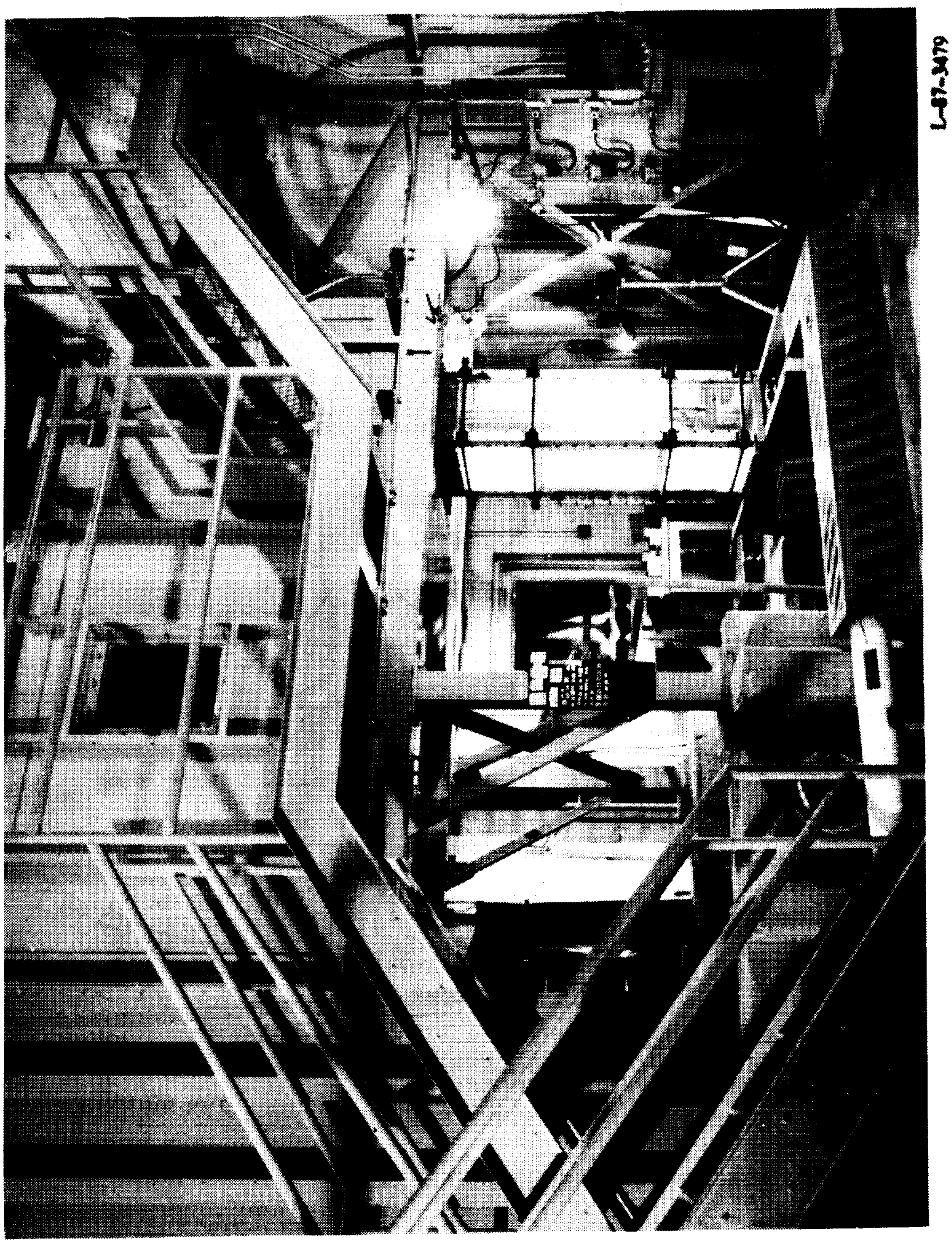

Figure XI-2. View of test section and dye reservoir panel with control console.

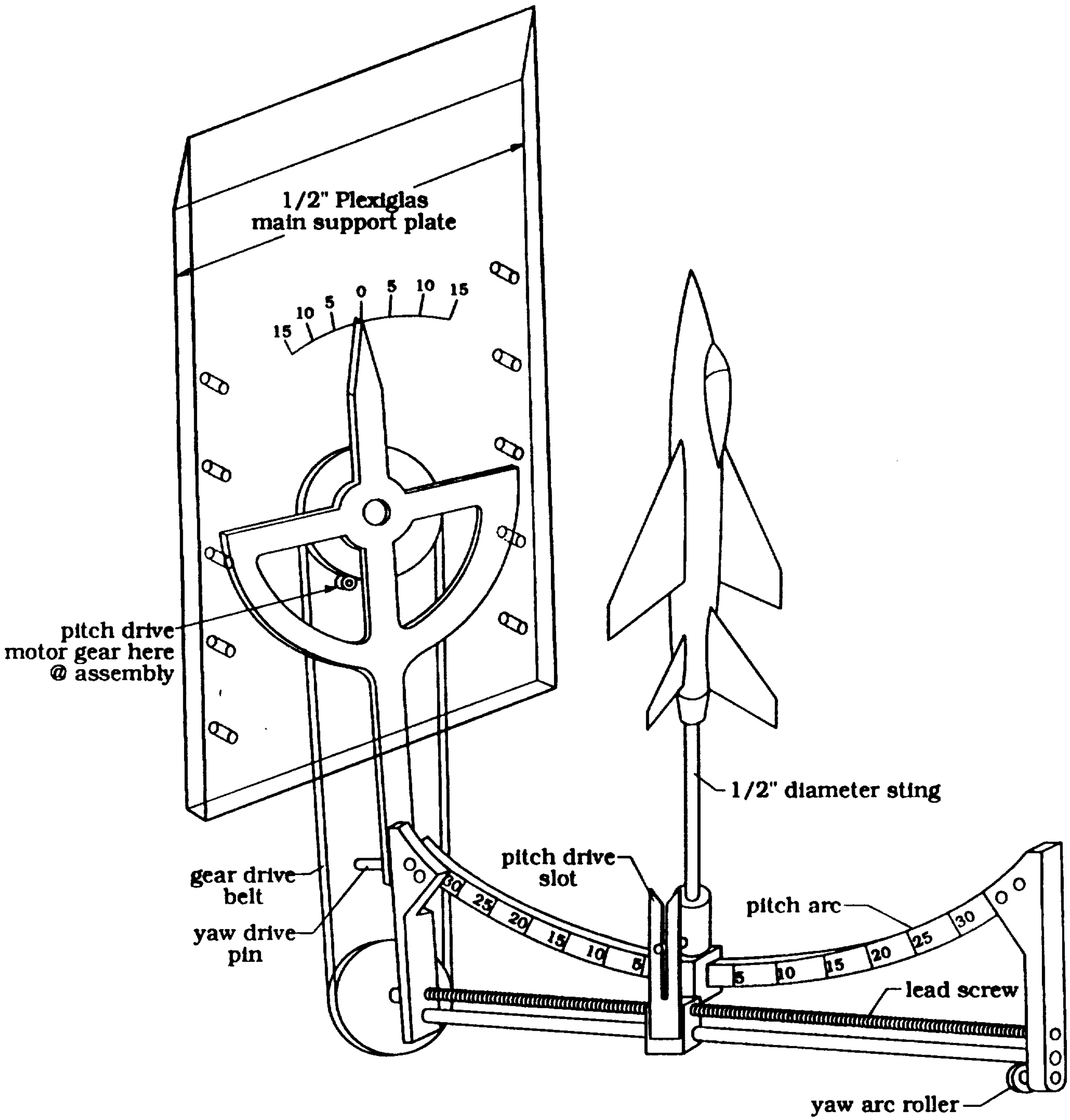

Figure XI-3. Sketch of model installed on AOA/AOY mechanism showing important parts.

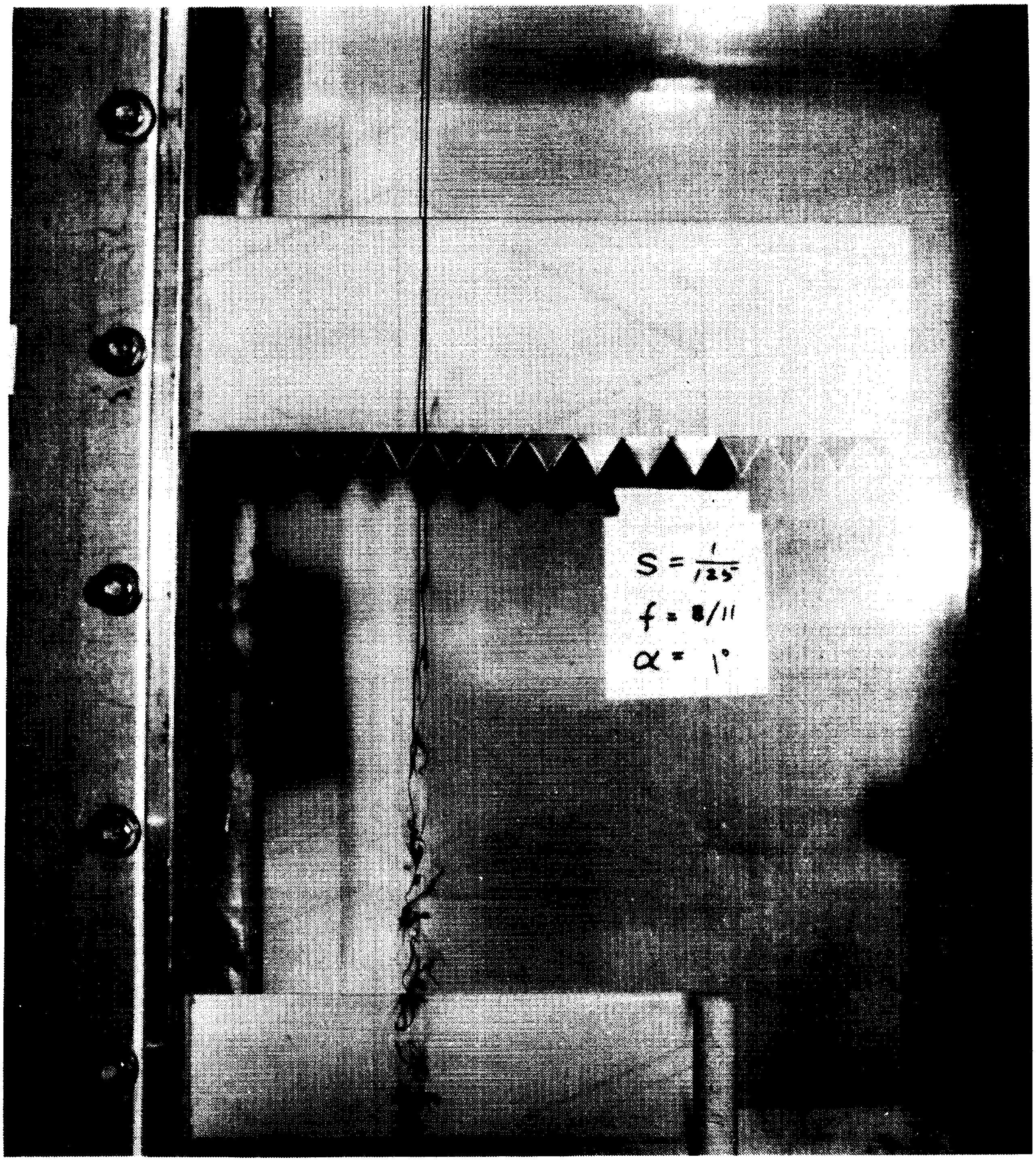

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Figure XI-4. Sting-mounted airfoil support plate assembly.

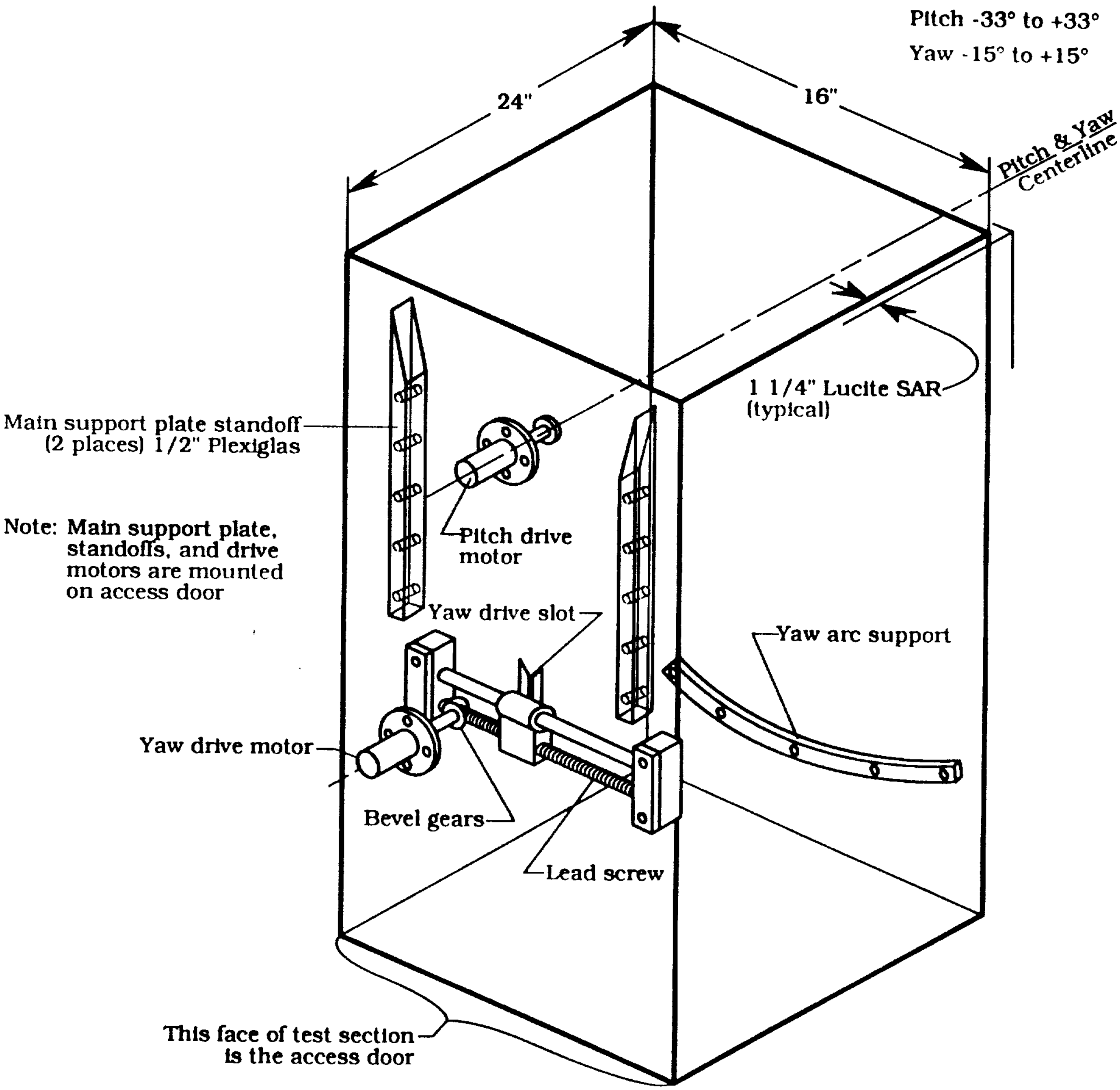

Figure XI-5. - View of test section showing AOA/access door interconnections

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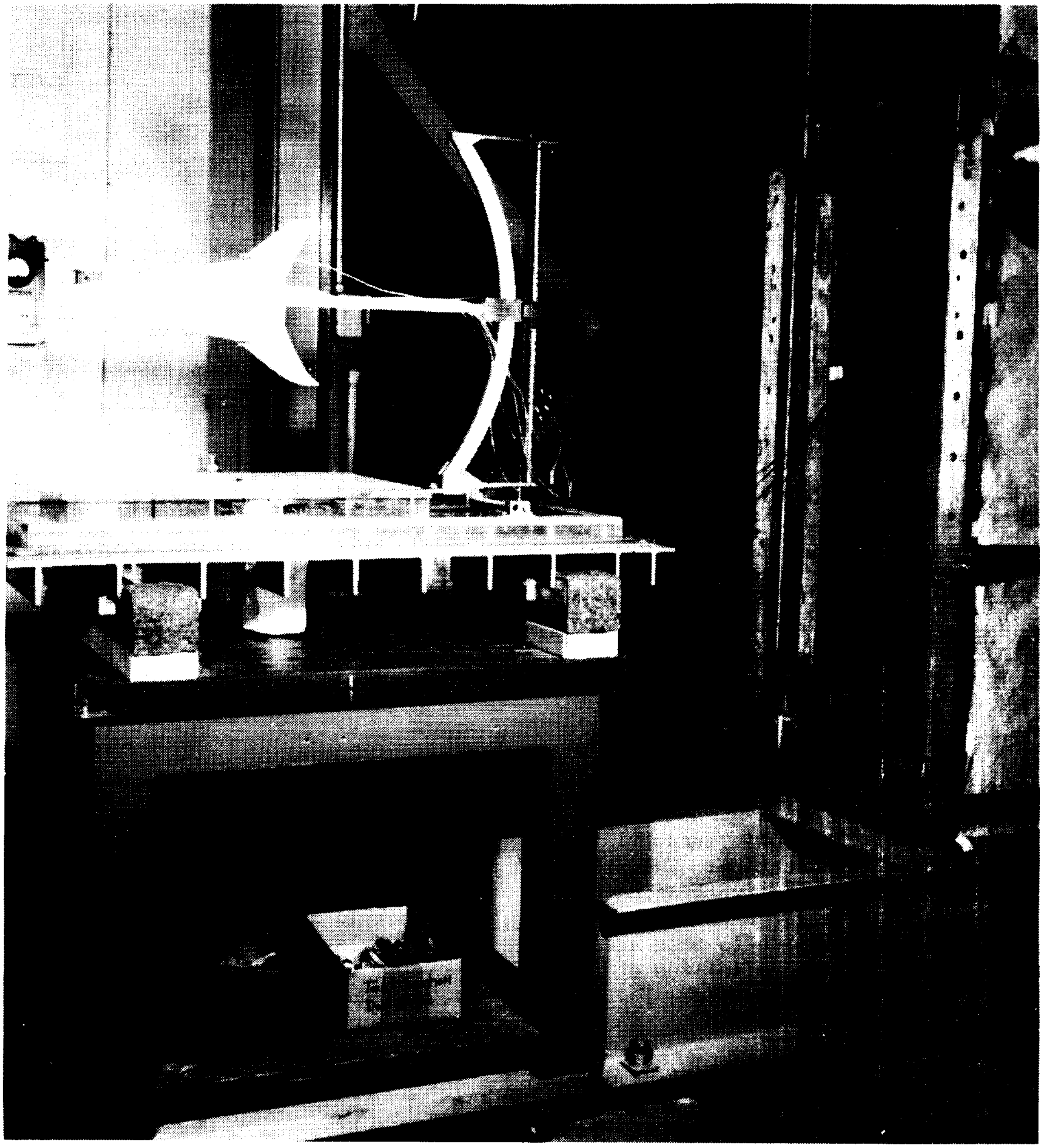

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Figure XI-6. Test section access door resting on work dolly.

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16. Abstract

The 16-Foot Transonic Tunnel is a single-return atmospheric wind tunnel having a slotted test section. The primary emphasis for research conducted in this facility is on the integration of the propulsion system into advanced aircraft concepts. The large test section size, 15.5 feet in diameter lends itself to conducting research in this area, where large models are required in order to provide adequate definition of the model geometry associated with the integration of the propulsion system. The nominal test Mach number range for this facility varies from 0.20 to 1.3.

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